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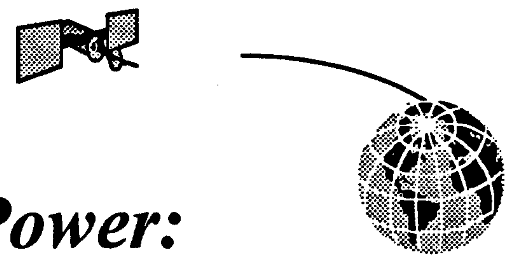

Space Propulsion and Power: Operational Effectiveness and Cost Study (OECS)

CHRISTOPHER A. FEUCHTER
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October 1996

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13. ABSTRACT *(Maximum 200 words)*

The performance of chemical upper stage rocket propulsion cannot be improved much in the near future. However, promising alternative solar- and nuclear-powered upper stages could be operational within ten years. This report provides a detailed discussion of these alternative upper stage concepts and their likely applications and performance. Included are descriptions of the methodology and the results of a cost-effectiveness analysis. The report also evaluates satellite electrical power generation because many of the concepts have propulsion and electrical power subsystems integrated with the satellite. We found frequent, significant performance and cost advantages for the alternative technologies compared with today's baseline. These technologies would make it possible to place the same payload mass into operational orbit with a smaller, less costly booster or to place more payload mass into orbit with the original booster. Off-setting these advantages are low thrust levels and, consequently, longer orbital transfer times.

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PREFACE

The *Space Propulsion and Power: Operational Effectiveness and Cost Study (OECS)* was proposed by Capt Fred Kennedy of Phillips Laboratory (PL/VTP) in the fall of 1993. In discussions with the Office of Aerospace Studies (OAS), the initial OECS scope was enlarged from an investigation of a single innovative upper stage propulsion and electrical power concept to include all potentially competitive contemporary concepts. OECS funding became available from PL/VTP in June of 1994, and the Air Force propulsion, development planning, and operational communities found the resources and desire to make the study a reality. Under OAS lead, the OECS kickoff took place that month.

While direct Air Force support was crucial to OECS success, the contractual participation of the Rocketdyne Division of Rockwell International led by Mr. Mike North and the support of Mr. Glen Law and others of the Aerospace Corporation were indispensable. A separate list of OECS participants is included in this volume. The OECS was truly a team effort, which OAS was privileged to coordinate.

We desired to be thorough in documenting the OECS, as the value of poorly documented analyses quickly fades. Because of this, some may find the level of detail included to be daunting. For those who want nothing but the "answers," Chapter 8 provides a concise, stand-alone summary. For those wanting more background, we suggest the introduction in Chapter 1.

A primary goal of the OECS was to provide reliable technology-planning guidance. Success in achieving the goal was ensured by employing:

- Advocate-sponsored technology concepts
- One methodology across all technologies
- Open decision making
- Advocate review of study inputs and outputs
- Consistent effectiveness and cost-effectiveness comparisons

These circumstances did not come cheaply, either in execution time or manpower. But sound analysis of complex problems is never quick or cheap. Slightly more than a year was required from kickoff to final briefing, and this document has been in preparation many additional months. Coordination meetings were frequent and occasionally contentious, yet we persevered. And as should be the norm rather than the exception, the models and methodologies developed for the OECS are being applied to a new, more complex scenario—a scenario based on reusable orbital transfer vehicles (ROTVs).

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ABBREVIATIONS AND ACRONYMS

ADACS	attitude determination and control system
AFMC	Air Force Materiel Command
AFSPC	Air Force Space Command
AMTEC	alkali metal thermal-to-electric conversion
APSA	advanced photovoltaic solar array
ATP	authority to proceed
avg	average
BOL	beginning-of-life
C&DH	command and data handling
CBS	cost breakdown structure
CERs	cost estimating relationships
COEA	cost and operational effectiveness analysis
CONOPS	concept of operations
CPV	command pressure vessel
cryo	cryogenic
DDT&E	design, development, test, and evaluation
deg	degree
ΔV	change in velocity
DOD	Department of Defense
DSCS	Defense Satellite Communication System
DSP	Defense Support Program
EELV	Evolved Expendable Launch Vehicle
EMD	engineering and manufacturing development
EOL	end-of-life
est.	estimated
FLTSATCOM	Fleet Satellite Communication System
FO	functional objective
FOC	full operational capability
FSD	full scale development
G&A	general and administrative
GaAs	gallium arsenide
GAP	generalized availability program
GEO	geosynchronous Earth orbit
GPS	Global Positioning System
GSE	ground support equipment
GTO	geosynchronous transfer orbit
HEO	highly eccentric orbitP
H_2	hydrogen
IA&T	integration assembly and test

IME	integrated modular engine
IOC	initial operational capability
I_{sp}	specific impulse
K	Kelvin
kg	kilogram
km	kilometer
kW	kilowatt
kW/m^2	kilowatts per square meter
lb	pounds
lbf	pound force
lbm	pound mass
LCC	life cycle cost
LEO	low Earth orbit
LH_2	liquid hydrogen
LLV3	Lockheed Launch Vehicle 3
LO_2	liquid oxygen
m	meter
MEO	mid-Earth orbit
MLR	multiple linear regression
MMD	mean mission duration
MMH	monomethyl hydrazine
MNS	mission needs statement
MOE	measure of effectiveness
MOP	measure of performance
MPE	materiel process expense
MPL	mass of payload
m/s	meter per second
N	Newton
NB	nuclear bimodal
NLS	National Launch System
NOAA/GOES	National Oceanic and Atmospheric Administration/ Geostationary Operational Environment Satellite
NUS	no upper stage
NWODB	new way of doing business
NH_3	ammonia
NiH_2	nickel dihydride
N_2H_4	hydrazine
N_2O_4	nitrogen tetraoxide
O&S	operations and support
OCEM	OECS Cost/Engineering Model
OECS	Operational Effectiveness and Cost Study (this study)
ORM	operational reference mission

P_a	constellation availability
PL	Phillips Laboratory
PMAD	power management and distribution
PPL	power of payload
psi	pounds per square inch
P^3I	preplanned product improvement
RCS	reaction control system
RDT&E	research, development, test, and evaluation
RORSAT	radar ocean reconnaissance satellite
RSM	response surface methodology
RTG	radioisotope thermoelectric generator
s	second
SB	solar bimodal
SBIRS	Space-Based InfraRed System
SE&I	system engineering and integration
SPT	stationary plasma thruster
SRMU	solid rocket motor upgrade
SSTO	single stage to orbit
ST	solar thermal
STE	special test equipment
STPTS	Solar Thermal Propulsion Transfer Study
TES	thermal energy storage
TFU	theoretical first unit
TNS	total number of satellites
TRL	technology readiness level
TT&C	telemetry, tracking, and commanding
TT&C and C&DH	telemetry, tracking, and commanding plus command and data handling
UFO	UHF Follow On
UHF	UltraHigh Frequency
USCM 7	Unmanned Spacecraft Cost Model, Version 7
W/kg	watts per kilogram
WBS	work breakdown structure
WER	weight estimating relationship
W·hr/kg	watt-hour per kilogram
w/wo	with/without
yr	year

1. INTRODUCTION

1.1 BACKGROUND

For years the military and civilian space propulsion communities have investigated innovative upper stage and on-orbit satellite propulsion concepts as alternatives for existing postbooster propulsion. These technologies promise possible stepdown to a smaller booster—thus saving launch costs—or enhanced payload capability without stepping up to a larger booster. The drawbacks of the technologies are developmental costs, less responsiveness due to low thrust and thus longer transfer time, large propellant tank volumes, and some environmental and political concerns. The *Space Propulsion and Power: Operational Effectiveness and Cost Study (OECS)* compares these technologies to current practice and to one another for effectiveness, cost, and cost-effectiveness.

While the innovative technologies are at various levels of maturity, all probably could achieve flight demonstration within seven years and initial operational capability (IOC) within ten years given suitable development funding. Because four of the technologies provide electrical power as well as propulsion, electrical power is included as a primary component of the study.

The major OECS innovative propulsion and power technology combinations are:

- Advanced cryo (advanced cryogenic propulsion and photovoltaic power)
- Nuclear bimodal (nuclear thermal propulsion and thermoelectric power)
- Solar bimodal (solar thermal propulsion and thermionic power)
- Solar thermal (solar thermal propulsion and photovoltaic power)
- Nuclear electric (nuclear electric propulsion and thermionic power)
- Solar electric (solar electric propulsion and photovoltaic power)

Figure 1-1 organizes propulsion technologies under the headings "Chemical," "Solar," and "Nuclear." The advanced cryogenic innovative technology ("Advance Cryo") represents an incremental improvement on the baseline cryogenic technology ("Current Cryo"). The remaining innovative technologies are solar and nuclear. The OECS solar and nuclear electric thrusters are arcjets and ion engines. Arcjets are examples of electrothermal propulsion, and ion engines are examples of electrostatic propulsion.

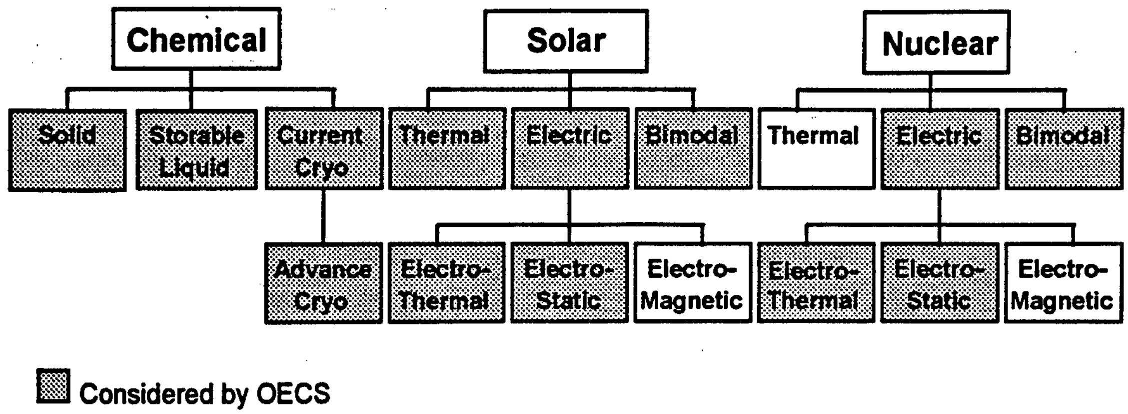


Figure 1-1. OECS propulsion technologies.

Nuclear thermal technology was not investigated because the Air Force program has been canceled and because the nuclear bimodal system employs nuclear thermal propulsion. The electromagnetic technologies were not considered because it is doubtful they could meet the OECS technology-demonstration and IOC requirements.

The baseline launch systems consist of existing operational systems: Delta II, Atlas IIAS, and Titan IV with the solid rocket motor upgrade (SRMU). All study technologies are assessed in a system-level context that includes aspects of ground operations, boosters, upper stages, and satellite.

The OECS is the most ambitious of three related studies funded by Phillips Laboratory's Bimodal Program Office (PL/VTP). The others are the Launch Vehicle Step-Down Study, conducted by W. J. Schaffer Associates, and the On-Orbit Asset Management Study, conducted by Lockheed Martin Astronautics. The Launch Vehicle Step-Down Study assesses the impact on launch vehicles of using the OECS innovative technologies to move payloads from existing launch vehicles to smaller, less expensive ones. The goal of the On-Orbit Asset Management Study is to investigate the novel on-orbit operational advantages of the innovative technologies.

Much of the OECS methodology is based on the *Solar Electric Propulsion Assessment* (Chan et al.) and the *Comprehensive On-Orbit Maintenance Assessment* (Feuchter et al., 1989).

1.2 GOALS AND SCOPE

The primary OECS goal is to provide as accurate a comparison of the capabilities, cost, and cost-effectiveness of the innovative technologies as resources and current

1.2 GOALS AND SCOPE

The primary OECS goal is to provide as accurate a comparison of the capabilities, cost, and cost-effectiveness of the innovative technologies as resources and current knowledge permit. This goal was pursued by sharing all pertinent technology and cost data among all participants, employing a common analytical methodology, and generating as consistent a set of cost-estimating relationships (CERs) as possible.

The OECS focuses on DOD missions. The study examines three propulsion tasks: orbital transfer to final mission orbit, on-orbit stationkeeping, and on-orbit maneuvering to new orbits. These tasks are referred to throughout this report as *lift*, *hold*, and *move*, respectively. Satellite electrical *power* requirements are also assessed because some OECS technologies use the same energy source to provide propulsion and electrical power.

1.3 STUDY TEAM

The OECS brought together the potential technology user, Air Force Space Command (HQ AFSPC/XPX); the space systems acquirer, Space and Missile Systems Center (SMC/XRT); the propulsion technology developers within Air Force Materiel Command, Phillips Laboratory (PL/VTP and OL-AC PL/RK); and major elements of the space analytical community. The Office of Aerospace Studies (AFMC OAS/DRA) provided the study leadership. The Aerospace Corporation and Rocketdyne, a division of Rockwell International, provided technical and cost support.

1.4 STUDY PANELS

Five panels were established: mission/threat, technology/concepts, analysis, cost, and policy/safety. Each major participating organization chaired at least one panel. The Aerospace Corporation provided system design/sizing and cost support. Rocketdyne provided a consistently derived set of CERs and a single-source familiarity with all technologies.

Figure 1-2 shows the interactions among the panels. The width of the arrows indicates the relative degree of interactions. The policy and safety panel is shown in the background to indicate its results provided ancillary information as opposed to influencing technical comparisons. Primary participants in each panel are listed beginning with the organizational panel chair. The design sizing model occupies a central place on the figure, as it did in the study. This model incorporates all technologies and provides a consistent set of upper stage and satellite designs.

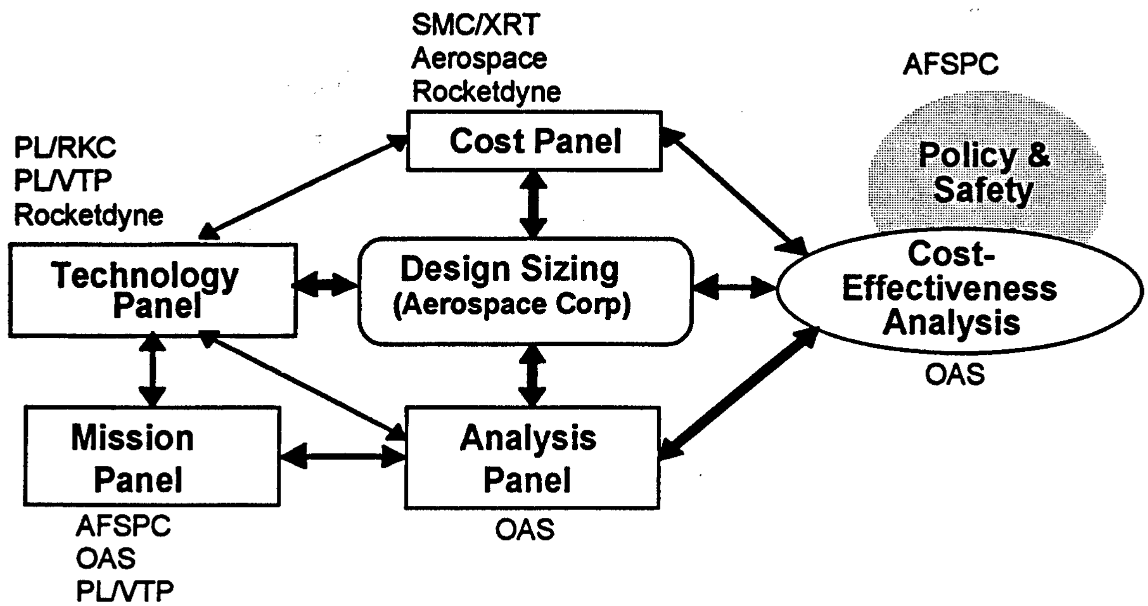
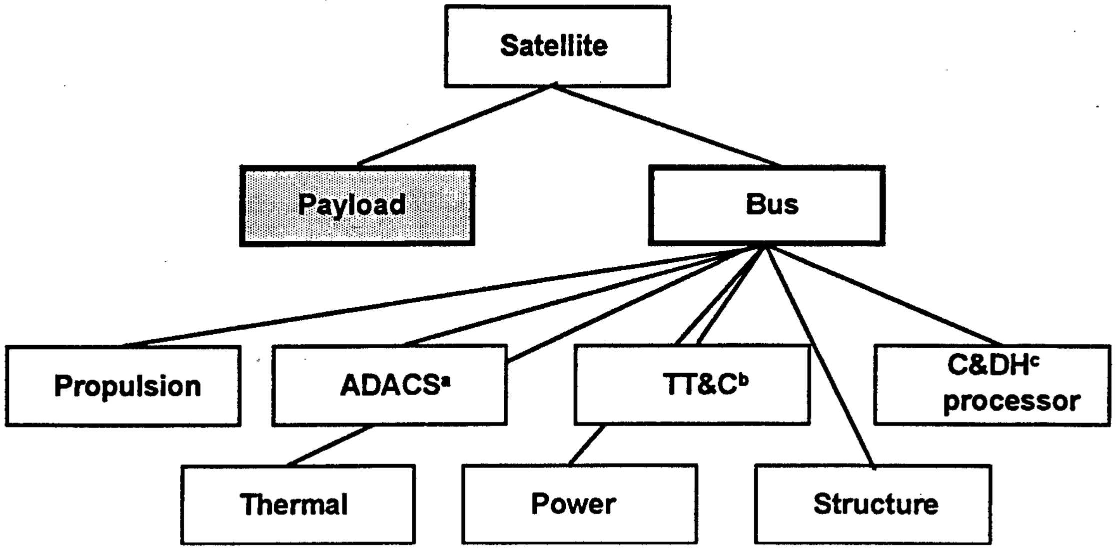


Figure 1-2. Primary relationships between OECS panels.

1.5 COMPARISONS OF TECHNOLOGY PERFORMANCE

Because of the diverse OECS propulsion technologies, a broad perspective of the term upper stage has been adopted in this report. Some OECS upper stages separate from the satellite and others are integral to the satellite. This diversity makes it impossible to make even a rough comparison of technology performances by citing total mass on orbit. For comparable satellite performance, the total mass on orbit of one technology may represent a satellite separate from its upper stage (as is done today), while another technology may include a massive nuclear reactor or solar mirrors that remain with the satellite.

Homogeneity in comparisons is reestablished by considering on-orbit payload mass and payload electric power in place of total mass on orbit. Figure 1-3 uses the terms satellite, payload, and bus to emphasize the nature of this paradigm shift. The satellite (total mass on orbit) is composed of the payload and bus. The bus supports the payload. The payload is the mission equipment—the reason for launching the satellite. We have shaded the payload block to emphasis the paradigm shift. The bus consists of seven



[a] Attitude determination and control system
[b] Telemetry, tracking, and commanding
[c] Command and data handling

Figure 1-3. Satellite organization.

subsystems, including on-orbit propulsion and electrical power. Bus subsystems are discussed in Chapter 6 and Appendix B.

1.6 GROUND RULES

Several ground rules were proposed and adopted during the study. The most significant are outlined below.

Ten-Year Initial Operational Capability (IOC): Only technologies with reasonable potential to achieve a flight demonstration in seven years and an initial operational capability within ten years were considered. This does not mean that all the technologies have the same developmental risk.

Existing Boosters: The OECS is not a booster study; rather, it is a comparison of upper stage lift, on-orbit hold and move propulsion, and electrical power technologies. Therefore, only existing boosters will be used. Consideration of incremental changes to existing launch vehicles or new launch vehicles would change the details of the results but would not change the relative capabilities and costs of the innovative technologies.

Technology Limitations: Known, quantifiable technology limitations (e.g., thruster lifetime, radiation degradation of photovoltaic cells) are accounted for in propulsion and electrical power designs. Less understood or potential limitations are discussed qualitatively (see Chapters 3 and 5).

Photovoltaics: The study assumes that photovoltaic technology advancement is the same for all applications, including baseline applications. This provides the fairest comparison, as it is reasonable to expect photovoltaic technology to be continually advancing independent of the development of any innovative technology.

Equal Reliability: All launch vehicles are assumed to be equally reliable. Known upper stage limitations, such as limited thruster lifetime or radiation-induced photovoltaic degradation, have been allowed for in the system designs and their projected employments. These limitations are discussed in Chapter 3 and Appendix B.

Innovative Technology/Payload Interaction: Each innovative technology will influence payload design in some manner. The OECS assumes that none of these interactions significantly impacts the use of the technology (known impacts are factored into system designs, e.g., radiation shielding for nuclear systems). Investigating this assumption is beyond the scope of this study.

Upper Stage Scaleability: With the exceptions of the advanced cryogenic and nuclear bimodal technologies, innovative upper stage designs are scaled (sized) exactly to the mission, payload mass, and payload electrical power. Advanced cryogenic has three designs: one each for Delta, Atlas, and Titan. Nuclear bimodal has a singular reactor design. Scaleability is discussed in more detail in Chapter 3 and Chapter 4.

Innovative Employment: The OECS scenarios or operational reference missions (ORMs) were selected because they represent known useful mission orbits. The structure of the study encourages independent consideration of how increased payload mass or electrical power on-orbit might be used. However identifying new applications is not a study goal.

Fairing Volumes: The OECS has roughly estimated the necessary fairing volumes associated with each combination of technology, mission, launch vehicle, and payload. Potential problems with fairing volume have been noted, but these problems do not restrict estimates of technology performance.

1.7 SCHEDULE

The OECS kickoff meeting occurred in June of 1994. The final briefing was given initially in July of 1995.

1.8 REPORT OVERVIEW

The study team identified six critical steps in accomplishing the OECS. These steps are listed below with a brief description of the chapters in which they are addressed.

1. *Examine a comprehensive range of potential operational scenarios.*
   Chapter 2 discusses potential uses for the innovative technologies and develops potential operational scenarios.

2. *Seek a uniform level of optimism in technology assumptions.*
   Chapter 3 gives an overview of each technology.

3. *Identify a comprehensive range of realistic technology combinations,*
   and
4. *Employ technologies correctly.*
   Chapter 4 presents an overview of the analysis along with details of many issues critical to understanding the analysis.

5. *Employ effectiveness and cost-effectiveness methodologies that are independent of the technology.*
   Chapter 5 details the methodology and results of the effectiveness analysis.

6. *Strive for consistent cost assumptions.*
   Chapter 6 describes the methodology and results of the cost analysis.

The main body of the report concludes with Chapter 7, which presents the results of the cost-effectiveness analysis, and Chapter 8, which is, in effect, an executive summary.

This document also contains five appendices: Appendix A discusses the relationship between satellite mean mission duration and design life; Appendix B addresses the sizing relationships in the aerospace-design sizing model; Appendix C deals with the response surface metholology equations for estimating constellation availability and number of satellites bought; Appendix D presents the fairing volume calculations for the innovative technologies; and Appendix E has the acquisition cost estimates for establishing and maintaining a constellation for 15 years. A sixth appendix, Appendix F, is available in electronic format to US government agencies~~ and their contractors~~. Appendix F consists of a series of Microsoft Excel™ spreadsheets containing the cost-estimating relationships (CERs).

2. OPERATIONAL REFERENCE MISSIONS

2.1 INTRODUCTION

The OECS Mission Panel incorporated operational missions into the OECS through the use of *operational reference missions* (ORMs). The ORMs define generalized mission orbits representing one or more satellite constellations that are operational or potentially operational and militarily useful. The ORMs ensure a real world flavor to the study. They help us select appropriate satellite and constellation parameters, choose analysis measures, select methodologies, and cost the technologies.

The following list identifies essential technology-independent operational parameters for constellations and satellites. The first two parameters affect the number of satellites bought over the lifetime of the constellation. The last three are basic yardsticks for defining the performance of propulsion/power technologies.

- Constellation size
- Satellite mean life
- Satellite payload mass
- Satellite payload electrical power
- Satellite on-orbit mobility

The mission panel was originally constituted as the mission/threat panel. No hostile threat analysis was performed—the panel did not have the specialized knowledge to evaluate potential threats with respect to the technologies. This does not imply that threats will not vary with technologies or ORMs. Differences in technology hardness and exposure times clearly exist, even if they are not well quantified.

2.2 DEFINITION CRITERIA

There are several ways to group operational constellations into ORMs: by orbital parameters, payload lift ΔV (velocity change) requirements, electrical power needs, mission function, etc. The OECS ORMs are grouped by orbital parameters—primarily semimajor axis and inclination, with eccentricity a consideration in some situations. This decision was made because the mission orbit adequately describes the lift aspects of the study and because only a small number of satellite orbits have historically been useful given temporal and geometric operational requirements.

2.3 ORM OVERVIEW

The mission/threat panel selected four general ORM groups: geosynchronous Earth orbit (GEO), mid-Earth orbit (MEO), low Earth orbit (LEO), and highly eccentric orbit (HEO). Interplanetary missions were excluded since they consist primarily of civilian applications. Six ORMs were identified within these groups, as shown in Figure 2-1.

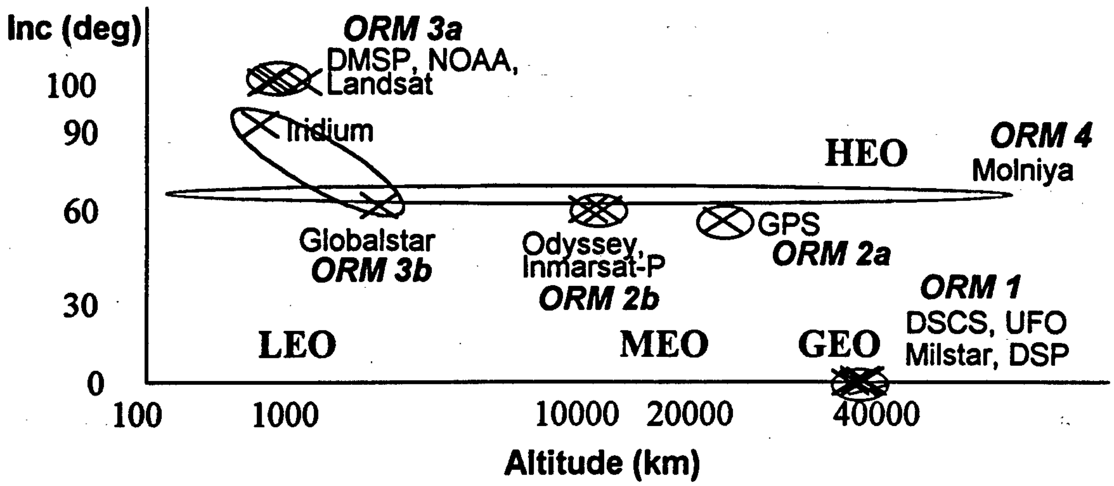


Figure 2-1. Satellite groupings into ORMs.

Most military missions occur in three ORMs:

- ORM 1: GEO
- ORM 2a: MEO-Global Positioning System (MEO-GPS)
- ORM 3a: LEO-Polar (LEO-P)

HEO (ORM 4) was included since it is a high energy, militarily useful orbit (as demonstrated by the Russian Molniya satellites). Two other ORMs—MEO-Low (ORM 2b) and LEO-Big (ORM 3b)—were initially included to encompass emerging non-GEO communications systems. These systems include Iridium®, Globalstar™, and Odyssey™. Limited OECS resources prevented the analysis of these ORMs.

To describe each ORM, the mission panel defined a number of characteristics about the orbit and the satellites likely to be found there. The orbit's semimajor axis, inclination, and eccentricity define lift requirements. In most cases, the orbit is that of a specific satellite type found in the ORM; in others, it represents a generic orbit. Stationkeeping (hold) can be determined from the satellite's orbit and attitude control. To characterize move requirements for each ORM, the mission/threat panel defined a "standard move" based on current satellite operations and/or capabilities. The panel also assessed the importance of lift, hold, and move propulsion requirements for each ORM, assigning a qualitative rating to each of high, moderate, or low. Constellation size, including on-orbit spares, rounds out the general description of the ORM.

A number of parameters further characterize the ORM. Ranges of these parameters describe the satellites and constellations typical of the ORM and define the parameter space for the cost-effectiveness analysis. The parameters are:

1. *Total Number of Satellites, TNS (#).* This parameter determines the size of the buy. Large buys introduce economies of scale because of production learning. In the OECS, total number of satellites is defined as the average number of

satellites required to establish and maintain a constellation for 15 years. This number is identical to the total number of launches. TNS ranges were estimated for each ORM using the GAP_PLUS simulation model. TNS is a function of constellation size, number of spare satellites, prelaunch preparation times, launch rates and reliabilities, satellite transfer and on-orbit checkout times, and satellite reliabilities.

2. *Mean Mission Duration, MMD (yr).* MMD is the average life of the satellite, and it can be related to the satellite's design life. The terminology for satellite reliability is not standardized. Under one interpretation, a satellite should not die before its design life (its average life can then exceed the design life through conservation and margin). Under another interpretation, design life is a truncation time—the satellite will not survive past its design life due to, for example, mechanical wearout or exhaustion of consumables. For the OECS, we have chosen the latter interpretation and have made additional assumptions to directly relate mean mission duration and design life. See Appendix A for a detailed discussion.

3. *Number of Standard Moves, N_{mov} (#).* This parameter defines a range of standard moves that a satellite in an ORM may have to perform. For ORMs with very limited move opportunities, a single value is given. US satellites typically have not had a large capability for on-orbit maneuvering—they are placed in an operational orbit and remain there. The obvious exceptions are satellites requiring end-of-life disposal to a nonoperational orbit and those moved (usually prior to or during conflict) to redistribute constellation coverage. For coverage redistribution, the amount of propellant expended is typically a function of how fast the maneuver is performed (degree/day), not the distance of the maneuver. All maneuvers are made at the expense of potential station-keeping propellant (hence possibly affecting satellite life). All moves are expressed in terms of impulsive ΔV.

4. *Payload Mass, MPL (kg).* Payload mass is the mass of the mission portion of a satellite. It excludes satellite structure and housekeeping functions; it includes mission hardware and all hardware directly associated with the payload, such as payload cooling systems (see Section 1.5).

5. *Payload Electrical Power, PPL (kW).* Payload electrical power refers to the sustained end-of-[design]-life (EOL) power requirement of the payload subsystem. EOL payload electrical power plus EOL bus electrical power defines total EOL satellite electrical power. EOL requirements are met by overdesigning satellite beginning-of-life (BOL) electrical power to account for time-related power system degradation (e.g., radiation-induced solar cell deterioration).

2.4 ORM MATRIX AND RATIONALE

The ORM matrix in Table 2-1 summarizes data for each ORM. A discussion of the matrix and supporting rationale is provided below.

2.4.1 ORM 1 (GEO)

Both military and civilian missions are performed in GEO: early warning, accomplished by the Defense Support Program (DSP) satellites; communications, accomplished by the Defense Satellite Communications System (DSCS), Milstar, Ultra-High Frequency (UHF) Follow-On (UFO), and many civil systems; and weather, performed by the National Oceanic and Atmospheric Administration's GOES (NOAA/GOES). GEO Constellation sizes typically range from three to five satellites. A minimum of three is required for complete equatorial coverage, and five provide some overlap and coverage to all but the Earth's polar regions. For example, NATO III is a four-satellite communications system and the Fleet Satellite Communications System (FLTSATCOM) is a five-satellite constellation (Muolo, pp. 92–93). On-orbit spares, if any, number one or two. (For OECS purposes, we use *constellation size* to represent the minimum number of satellites required to perform the mission.) Active spares may be used to increase constellation availability. AF Space Command does not differentiate between mission satellites and spares. The command includes both when referring to constellation size, while recognizing that not all satellites may be necessary to meet requirements.

GEO altitude is 35,786 km, which gives satellites a period of one sidereal day. According to information furnished by Glenn Law of Aerospace Corporation, the typical orbit is circular and equatorial, but some GEO satellites are slightly inclined when north-south stationkeeping has a high tolerance. For the OECS, the mission panel used a circular equatorial orbit. Lift ΔV requirements are high, from 4,285 m/s for impulsive maneuvers to 5,913 m/s for continuous, low-thrust electric propulsion. Tight stationkeeping tolerances, similar to DSCS, were assumed for GEO. These tolerances require 51.38 m/s ΔV per year for north-south stationkeeping and 5 m/s per year for east-west stationkeeping and attitude control. The standard GEO move is based on two maneuvers all GEO satellites are likely to perform: a slow move from a test location to an operational slot (1 deg per day) and a disposal move. The first requires about 5.7 m/s, and the latter about 20.06 m/s (assuming a DSCS III-like disposal)—for a total of 25.8 m/s. (See the sidebar "Satellite On-Orbit Maneuvers.")

System characteristics are based on a number of existing or planned GEO satellite constellations (Table 2-1). GAP_PLUS estimated the total number of satellites based on a constellation availability of 90%. Based on the relevant parameters, a range of 5–25 satellites was appropriate (see Chapter 5, MOE-1, for a detailed discussion). An orbital analysis determined the range of moves. The analysis was performed using standard GEO repositioning maneuvers: a satellite transfers to a higher or lower circular orbit, waits until it has reached its new position, and then transfers back to its original altitude. The intermediate circular orbit is necessary to avoid moving in and out of the GEO band.

Table 2-1. ORM Matrix

Operational Reference Mission (ORM)	GEO	MEO		LEO		HEO
	ORM 1	ORM 2a MEO-GPS	ORM 2b MEO-Comm	ORM 3a LEO-Polar	ORM 3b LEO-Big	ORM 4
General Description						
Missions	Communication, early warning, weather	Navigation	Communication	Weather, remote sensing	Communication, possible early warning	High latitude missions similar to GEO
Example Systems	DSCS III, GOES, Milstar, UFO	GPS	Odyssey, Inmarsat-P	DMSP, NOAA, Landsat	Globalstar, Iridium	Russian Molniya
Focus	Lift: High Hold: Mod Move: Mod	Lift: High Hold: Low Move: Low	Lift: High Hold: Low Move: Low	Lift: Low Hold: Low Move: Low	Lift: Low Hold: Low Move: Low	Lift: High Hold: Low Move: Low
Standard Orbit	alt = 35,786 km e = 0 i = 0°	alt = 20,200 km e = 0 i = 55°	alt = 12,000 km e = 0 i = 60°	alt = 850 km e = 0 i = 98.8°	alt = 1600 km e = 0 i = 50°	alt-p = 1000 km alt-a = 39,368 km i = 63.44°
Total Constellation (incl. spares)	3–5 (+0,1)	21 (+3)	12 (+3)	1–3 (+0)	74	2
Satellites Required for Mission	3–5	21	9–12	1–3	48, 66	2
Stationkeeping ΔV (m/s/yr)	51.38 (N-S) + 5	0.15	1.5	1.0	1.0	6.2
Standard Move ΔV (m/s)	25.8	27.17	30	20.0	250	95.0
System Characteristics						
Total Number of Satellites, TNS (#)	5–25	45–70	20–40	2–15	150–300	4–8
Mean Mission Duration, MMD (yr) [Equivalent Design Life (yr)]	5–14 [6–16.8]	8–12 [9.6–14.4]	8–12.5 [9.6–15]	4–7 [4.8–8.4]	4–7.5 [4.8–9]	5–10 [6–12]
Number Standard Moves, N_{mov} (#)	1–15	1–6	1–2	1–6	1–3	1–3
Payload Mass, MPL (kg)	200–2000	200–500	500–2000	500–1500	150–750	500–2000
Payload Electric Power, PPL (kW)	0.5–5	0.5–1.5	1–4	0.5–2	0.5–1.5	0.5–5

Satellite On-Orbit Maneuvers

There are several possible reasons for maneuvering a satellite in its mission orbit:

1. *Check-out.* Some satellites, typically those in GEO, are initially placed in a test location, checked-out, and then maneuvered to their operational location. This maneuver has low ΔV requirements.

2. *Disposal.* At the end of a satellite's life, the satellite is either moved into a disposal orbit or de-orbited. Typically, the disposal orbit is at a higher altitude. This type of maneuver is important for large constellations or in the crowded GEO band. The maneuver itself can be complex: GEO satellites, for example, typically transfer to a higher orbit using an elliptical transfer orbit, circularize, then repeat the process until the desired disposal orbit is reached (300 nm above GEO in the case of DSCS III). This complex maneuver is required since the amount of propellant remaining is uncertain. Circularizing at intermediate orbits ensures that if the full maneuver cannot be completed, the satellite will not be trapped in an elliptical orbit that intersects the GEO band.

3. *Crisis Coverage.* Satellites can be repositioned or constellations optimized to provide mission-essential coverage in times of war, crisis, or operational need. GEO satellites were repositioned in support of Desert Storm.

4. *Emergency Stabilization (Safing).* Propellant can be allocated to spin-up a tumbling satellite, allowing the satellite to rotate predictably. When spun down, the satellite can be reoriented. This is the case with GPS.

5. *On-orbit sparing.* A satellite would be placed in a storage orbit and then moved to replace a failed satellite.

6. *Evasion.* An evasive maneuver avoids a threat, whether hostile or natural. Threat avoidance requires tactical warning (i.e., knowledge that the threat is coming) and the ability to outmaneuver the threat. In the case of a long-term nuclear effect, there also would have to be a "safe" orbit that would permit meaningful operations. Since these conditions are difficult to meet, satellites typically enhance their survivability through increased hardening.

7. *Spoofing.* A spoofing maneuver would be used to avoid satellite orbital characterization. One could envision using this type of maneuver to cause uncertainty in estimating a satellite's fly-over time.

8. *Constellation dispersal.* Constellations with large numbers of satellites in an orbital plane may launch several satellites on a single launch vehicle. These maneuvers would disperse the satellites into their operational positions after being dropped off by the launch vehicle: the satellites would be in elliptical transfer orbits and individually circularize when they arrive at their operational locations.

9. *Constellation downsizing.* Through maneuvers, satellites would be able to position themselves where needed as they are needed, thereby decreasing the requirement for larger constellations.

Of these maneuvers, only the first four—check-out, disposal, crisis coverage (to some extent), and emergency stabilization—represent current requirements.

The upper limit of the chosen move range is based on a 60-deg/day move performed by a DSCS II satellite in support of Desert Storm, which nearly exhausted its on-board propellant (Newman). Most GEO satellites, however, only perform 1–3 deg/day moves (0.22–0.66 standard moves). Table 2-2 provides a summary and rationale for the chosen ranges of all the GEO parameters.

Table 2-2. ORM 1 (GEO): Typical System Characteristics and Their Translation to OECS Parametric Ranges

Characteristic	Examples	OECS Range/(Rationale)
Total # Satellites (TNS)	3-satellite constellation (w/wo spare) MMD ≈ 5 yr: TNS ≈ 12–15 MMD ≈ 14 yr: TNS ≈ 5–6 5-satellite constellation (w/wo spare) MMD ≈ 5 yr: TNS ≈ 20–23 MMD ≈ 14 yr: TNS ≈ 8–10	5–25 (reasonable range)
Mean Mission Duration (MMD), Design Life	DSCS III: 7-yr MMD, 10-yr design life UFO: 12.6-yr MMD,14-yr design life Intelsat 7A: 10.9-yr design life, 16-yr fuel Intelsat 8: 14–18-yr design life	5–14-yr MMD, 6–16.8-yr design life (covers present and near-term satellite reliabilities)
# Standard Moves	Standard Move (25.8 m/s) Disposal (20.06 m/s): 0.78 move Check-out (≈ 5.7 m/s): 0.22 move Additional moves (deg/day) ±1 (5.7 m/s): 0.22 move ±3 (17.0–17.2 m/s): 0.67 move ±5 (28.2–28.7 m/s): 1.10 moves ±10 (55.9–58.0 m/s): 2.25 moves ±60 (308–385 m/s): 12–15 moves	1–15 (every GEO satellite typically performs one standard move; 15 moves also includes one 60-deg/day transfer using avg. DV)
Payload Mass	DSCS III: 230 kg (500 lb) Milstar III: 2079 kg (est.) DirecTV: 1270 kg total satellite Intelsat 7A: 1748 kg total	200–2000 kg (encompasses range of satellites from DSCS III to newest planned Milstar)
Payload Power	DSCS III: 500 W Milstar III: 2.13 kW (est.) DirecTV: 4.07 kW (est.) Intelsat 7A: 3.75 kW	0.5–5 kW (encompasses range of DSCS III to advanced telecommunications satellites, plus 1 kW margin)

2.4.2 ORM 2a (MEO-GPS)

The Global Positioning System (GPS) is one of two ORMs identified at MEO. The GPS constellation consists of 24 satellites in six planes inclined at approximately 55 deg. Since only 21 satellites are required to provide three-dimensional position and time information for the entire Earth, one satellite in alternating planes functions as an active spare—hence our description of 21 (+3) satellites. The satellites are in circular, semi-synchronous (12-sidereal hour) orbits at an altitude of approximately 20,183 km.

Impulsive lift ΔV requirements are high: 3,512 m/s (3,813 m/s for continuous, low-thrust electric propulsion). On-orbit propulsion requirements, on the other hand, are relatively low. The orbital altitude and the nature of the mission do not require much stationkeeping: Aerospace Corporation estimates 0.15 m/s per year (Chao). Maneuvers are also minimal. Even though the Block IIR satellites will carry 100 kg of hydrazine propellant, GPS satellites are frequently retired with most of this propellant still on-board (Slokum). GPS satellites have at least two required maneuvers: disposal to at least 93 km (50 nm) above the GPS orbit and one or two safing maneuvers that spin up the satellite to 25 rpm in case of an emergency (Skokum). The GPS standard move is thus an aggregate of these two maneuvers: 27.17 m/s (6.73 m/s for disposal, 10.22 m/s per safing maneuver for two maneuvers). Although unlikely, should two satellites in the same plane fail, the two remaining satellites may be repositioned to improve constellation performance.

GPS system characteristics are based on estimated characteristics of the Block IIR and IIF satellites. (The IIR satellites are replacements for the current Block IIA satellites; IIF is the follow-on to the IIR.) The total number of satellites to populate and maintain a GPS-like constellation is estimated to be between 45–70 satellites with very high corresponding constellation availabilities. The maneuver range is based on estimated IIR on-orbit capabilities and does not necessarily reflect operational requirements. The other parameters, as described in Table 2-3, cover a wide range of values. While the upper limits on payload mass and electrical power may seem high given the likelihood IIF satellites will be smaller than IIR satellites, the values are still reasonable for the purposes of the study.

2.4.3 ORM 2b (MEO-Comm)

The second ORM at MEO is at a lower altitude than GPS, having a 6-hr period instead of GPS's 12-hr period. Although identified by the mission/threat panel as interesting, resources were not available to include it in the analysis process. No systems currently occupy this region. However two companies are proposing communications systems: TRW-Teleglobe's Odyssey system and the International Maritime Satellite (INMARSAT) Organization's INMARSAT-P system (now INMARSAT ICO [Inclined Circular Orbit]). The mission/threat panel decided to break out this ORM from GPS for two reasons: (1) the orbit is lower than that of GPS; and (2) the orbit sits within the Van Allen belts, and the harsher environment will have a significant impact on photovoltaic power systems and other electric components. The MEO-Comm orbit is based on Odyssey's 12,000 km altitude and 50-deg inclination in three orbital planes.

Table 2-3. ORM 2a (MEO-GPS): Typical System Characteristics and Their Translation to OECS Parametric Ranges

Characteristic	Examples	OECS Range/(Rationale)
Total # Satellites (TNS)	21-satellite constellation with 3 spares MMD ≈ 7 yr: TNS ≈ 72 MMD ≈ 10 yr: TNS ≈ 57 (GPS requires about 48 satellites per block)	45–70 (reasonable range)
Mean Mission Duration (MMD), Design Life	IIR: MMD approx. 7.5 yr, wearout at approx 10 yr (est.)	8–12-yr MMD, 9.6–14.4-yr design life (low end of range reflects GPS IIR; high end reflects reasonable advances in solar cell and battery life)
# Standard Moves	Standard Move (27.16 m/s) Disposal (6.73 m/s): 0.25 move Safing (2 at 10.22 m/s): 0.75 move Additional moves (deg/day) ±1 (3.6 m/s): 0.13 move ±5 (17.9 m/s): 0.66 move Two-satellite failure scenario, within 4 days (80.54 m/s): 3 moves 1 day (322.5 m/s): 11.9 moves	1–6 (GPS has an estimated allocation of approximately one standard move, and Block IIR has an estimated capability for about six moves)
Payload Mass	IIR: 360 kg payload IIF: approx. 200–250 kg (est.)	200–500 kg (encompasses current GPS plans plus additional margin for growth)
Payload Power	IIR: 850 W (est. 680–760 W for payload) IIF: lower than IIR Estimate 1000–1200 W total power growth for a IIR-equivalent satellite (800–1080 W for payload)	0.5–1.5 kW (encompasses current GPS plans plus reasonable margin for growth)

Odyssey is comprised of 12 satellites arranged in three planes. According to TRW, 9 satellites are required to provide worldwide coverage; 12 satellites are required to provide dual worldwide coverage. The MEO-Comm orbit is based on Odyssey's 12,000-km altitude and 50-deg inclination.

Lift requirements to this orbit are slightly lower than for GPS. Stationkeeping is also low. For example, Odyssey is planning 1.5 m/s per year (Pritchett). The standard move is 30 m/s, which is based on Odyssey's 15 kg of on-orbit maneuver propellant (Pritchett). Table 2-4 summarizes the ORM's characteristics.

Table 2-4. ORM 2b (MEO-Comm): Typical System Characteristics and Their Translation to OECS Parametric Ranges

Characteristic	Examples	OECS Range/(Rationale)
Total # Satellites (TNS)	12-satellite constellation with 3 spares MMD ≈ 8 yr: TNS ≈ 39 satellites MMD ≈ 14 yr: TNS ≈ 22 satellites	20–40 (reasonable range)
Mean Mission Duration (MMD), Design Life	Odyssey: 12-yr MMD, 15-yr design life Inmarsat-P: 10-year life	8–12.5-yr MMD, 9.6–15-yr design life (encompasses range)
# Standard Moves	Standard move (30 m/s) Disposal to 90 km (12 m/s): 0.4 move	1–2 (approximate capability of Odessey system)
Payload Mass	Odyssey: 3800 kg total (wet), 1135 kg BOL Inmarsat-P: 1244 kg BOL	500–2000 kg (encompasses range including growth)
Payload Power	Odyssey: approx. 3 kW payload, 3.5 kW total EOL (another source lists 1800 W total) Inmarsat-P: 3760 W total	1–4 kW (encompasses range including growth)

2.4.4 ORM 3a (LEO-Polar)

The first LEO ORM is LEO-Polar (LEO-P), representing a circular near-polar orbit. Most of the unmanned LEO missions fall into this ORM. Low-inclination LEO missions tend to be manned or involved experimental payloads (Thompson, pp. 54–59). The LEO-P ORM is sun-synchronous, meaning that the inclination is set to a special value (slightly larger than 90 deg) that takes advantage of orbital precession to keep the orbit's orientation constant with respect to the sun. Thus, the orbit periodically passes over a point on the earth at the same local time. LEO-P missions include: weather, accomplished by satellites such as the Defense Meteorological Satellite Program (DMSP); and earth observation, accomplished by the US Landsat and French SPOT satellites. Constellations tend to be small—one to three satellites. We assume there are no on-orbit spares.

The LEO-P orbit is arbitrarily based on the DMSP system. This orbit is circular at an altitude of 850 km and an inclination of 98.7 deg. Lift ΔV requirements are low compared to the other OECS orbits. Stationkeeping requirements are also low. Atmospheric drag is almost negligible at 850 km for LEO mission durations and typical spacecraft sizes (Kechichian). Solar pressure perturbations are minor. Solar attraction does shift the orbit's inclination, affecting its sun-synchronous property (Chobotov, pp. 250–251). The effect can be minimized by biasing the satellite's initial orbit (as is done by DMSP) or by modifying altitude (Chao). Maintaining satellite phasing becomes a priority for satellite constellations, but this can be done by adjusting altitude (Chao). Overall, requirements are very low—less than 1 m/s per year for the DMSP/NOAA follow-on. Based on this information, the mission/threat panel choose 1 m/s per year for the ORM.

Maneuver requirements are also low. (DMSP carries very little on-board propellant.) Two types of maneuvers seem possible for LEO-P: disposal and coverage changes. Since the ΔV associated with quickly changing coverage patterns can be very high, the mission/threat panel decided to confine LEO-P maneuvers to disposal only. LEO satellites eventually reenter the atmosphere, and a disposal maneuver would lower the satellite's orbit to facilitate reentry. The standard move was defined as 20 m/s, which is based on the approximate total ΔV capabilities of a DMSP Block 5D3.

Table 2-5 summarizes the characteristics of LEO-P satellites used for the study. Most are reasonable ranges based on current and planned systems. The range for the number of moves encompasses the total on-orbit ΔV capabilities of a number of satellites. Since sun-synchronous satellites vary in altitude, some satellites need more stationkeeping for drag make-up. Thus, the six moves can represent a high-altitude satellite with lots of maneuverability (or a reentry facilitating maneuver) or a lower-altitude satellite with higher stationkeeping than our 1 m/s/yr.

2.4.5 ORM 3b (LEO-Big)

The second LEO ORM, LEO-Big, captures a number of emerging missions at LEO that will be accomplished by large constellations (in the communications arena, the so-called Big LEO missions). These systems include Motorola's Iridium system (involving 66 satellites) and Loral-Qualcomm's Globalstar concept (involving 48 satellites plus 8 on-orbit spares). In addition, a Space-Based Infrared System (SBIRS) may have a LEO component involving 16 to 24 satellites according to an early TRW concept (Weber, pp. 3, 16). As was the case for MEO-Comm, limited OECS resources prevented the inclusion of LEO-Big in the analysis process.

LEO-Big is very similar to LEO-P except for constellation size. Iridium is roughly at our LEO-P altitude. Globalstar places its satellites in a 900-km circular orbit before using on-board propellant to raise them to 1400 km. The Globalstar strategy is to leave spares in the lower orbit and raise them as needed. The lower altitude experiences perturbations which position the spare in the necessary orbital plane. The standard move is based on the ΔV required to move the satellite to 1400 km (like Globalstar). The range of moves for Globalstar is based on the amount of on-board propellant.

Table 2-5. ORM 3a (LEO-Polar): Typical System Characteristics and Their Translation to OECS Parametric Ranges

Characteristic	Examples	OECS Range/(Rationale)
Total # Satellites (TNS)	1-satellite constellation (no spare) MMD ≈ 5 yr: TNS ≈ 4 3 satellites (no spares) MMD ≈ 3 yr: TNS ≈ 12	2–15 (reasonable range)
Mean Mission Duration (MMD), Design Life	NOAA/DMSP: 7-yr design life SPOT 4: 5-yr design life, approx. 4-yr MMD	4–7-yr MMD, 4.8–8.4-yr: design life (reasonable range)
# Standard Moves	Standard Move (20 m/s) NOAA/DMSP (23.4 m/s total ΔV, 6.3 m/s for stationkeeping) Approx. total ΔV capability (est.) Landsat 6 (60 m/s): SPOT 4 (120 m/s): Disposal to 450 km perigee (106 m/s):	1–6 (reflects range of satellite total capabilities; adequate for de-orbital disposal or to account for additional stationkeeping for lower altitude satellites)
Payload Mass	NOAA/DMSP: 888-kg payload mass Landsat 6: 1650-kg dry mass (approx.) SPOT 4: 1400-kg payload, 2500-kg total mass Radarsat: 1366 kg	500–1500 (reasonable range)
Payload Power	NOAA/DMSP: 1.06-kW payload power Landsat 6: 1.26 kW total (EOL) SPOT 4: 2.2 kW total (EOL) Radarsat: 3 kW total	0.5–2.0 kW (reasonable range)

The ORM's orbit is based on the Globalstar concept due to its higher altitude and therefore somewhat higher lift ΔV—to approximately 1400 km vs Iridium's 780 km. However, no suboperational storage orbit was considered. Orbital inclination is 52 deg (vs 86.4 for Iridium). Stationkeeping requirements are low. The most important aspect of stationkeeping is to maintain satellite phasing. Iridium plans to de-orbit its satellites. The mission/threat panel choose 1 m/s/yr as the requirement based on the LEO-P requirement. Table 2-6 summarizes the characteristics of the LEO-Big satellites used in this study.

Table 2-6. ORM 3b (LEO-Big): Typical System Characteristics and Their Translation to OECS Parametric Ranges

Characteristic	Examples	OECS Range/Rationale
Total # Satellites (TNS)	66-satellite constellation with 7 spares MMD ≈ 5 yr: TNS ≈ 270–290 MMD ≈ 12 yr: TNS ≈ 170	150–300 (reasonable range)
Mean Mission Duration (MMD), Design Life	Globalstar: 7.5-yr lifetime Iridium: 5–7-yr lifetime	4–7.5-yrs MMD, 4.8–9-yr design life (reasonable range)
# Standard Moves	Standard move (250 m/s) Raise orbit 700–1400 km (250 m/s): 1 move De-orbit from 1400 km (750 m/s): 3 moves De-orbit from 780 km (450 m/s): 1.8 moves	1–3 (reasonable range)
Payload Mass	Globalstar: 222-kg total dry mass, 400-kg wet Iridium: 700-kg wet mass	150–750 kg (encompasses estimated payload masses plus growth)
Payload Power	Globalstar: 875 W total satellite (peak) Iridium: 1200 W total	0.5–1.5 kW (encompasses estimated payload powers plus growth)

2.4.6 ORM 4 (HEO)

The Highly Eccentric Orbit (HEO), or the "Molniya" orbit, is a specialized orbit. It ". . . was devised by the USSR to procure features of a geosynchronous orbit with better coverage of the northern latitudes" (Brown, p. 90). The Russians have used this orbit for domestic communications and early warning (Thompson, pp. 54–59).

Like GPS, HEO has a period of 12 sidereal hours and thus a semimajor axis of about 26,562 km. Unlike GPS, the orbit is very eccentric: typical values range from 0.64–0.74. Perigee altitudes vary but must be high enough to preclude reentry during mission life. According to Thompson (pp. 54–59), Russian Molniya perigees averaged about 650 km in 1993. The orbit's inclination is typically 63.44 deg to minimize the effect of orbital perturbations and keep the argument of perigee close to 270 deg. These orbital parameters allow a HEO satellite to remain in view of the northern hemisphere during most of its orbit. In fact, two properly placed satellites will continuously view 55–60% of the hemisphere centered on the North Pole (Chobotov, pp. 288–291).

For the purposes of the study, we have chosen a 1000 × 39,464 km orbit (eccentricity 0.722). (The average Russian perigee was considered to be too low.) This orbit avoids most LEO satellites, and the final analysis results are not significantly different. Orbital inclination is fixed at 63.44 deg. Constellation size is set at two satellites with no spares. Properly maintaining the phasing of the constellation requires about 3.1 m/s/yr according to analysis by Aerospace Corporation (Chao). We have doubled that requirement to take into account attitude control and other requirements for a total of 6.2 m/s/yr.

If polar coverage is the desired mission of HEO satellites, there are few reasons for on-orbit maneuvering. Since maneuvers would destroy the phasing required for continuous coverage, EOL disposal is the only reasonable maneuver. Aerospace Corp. has just begun looking at the problem of HEO satellite disposal (Chao). A simple approximation suggested by Wertz and Larson is to lower perigee to 0 km altitude (pp. 155–156). This maneuver would require 95 m/s and is the basis of the HEO standard move. One problem with this approach—or simply letting the orbit decay—is that the reentry angle would be too steep to ensure complete satellite burnup (Chao). Lowering apogee may help. Changing to a 600 × 26,863 km orbit (lower perigee by 400 km to increase drag and lower apogee by 12500 km) would require 275 m/s or about three standard moves. Thus, the range of 1–3 moves would cover a range of disposal options, including moving the old satellite out of the way of an incoming new one.

The remaining system characteristics are addressed in Table 2-7. Payload mass and power are based on GEO satellites since the Russians have used the orbit for similar purposes. However a HEO satellite would require additional shielding (Law). The MMD was capped at 10 years because of continued travel through the Van Allen radiation belts, but this cap does not imply an engineering limit on HEO satellites.

Table 2-7. ORM 4 (HEO): Typical System Characteristics and Their Translation to OECS Parametric Ranges

Characteristic	Examples	OECS Range/(Rationale)
Total # Satellites (TNS)	2-satellite constellation MMD ≈ 5 yr: TNS ≈ 8 MMD ≈ 14 yr: TNS ≈ 4	4–8 (reasonable range)
Mean Mission Duration (MMD), Design Life	GEO satellites: Comm satellites: up to 14-yr MMD	5–10-yr MMD, 6–12-yr design life (range similar to GEO satellites, capped at 10 yr to take into account Van Allen radiation)
# Standard Moves	Standard move: 95 m/s Simple reentry (95 m/s): 1 move Disposal reducing perigee and apogee - 100 and 10,000 km below HEO (190 m/s): 2 moves - 400 and 12,500 km below HEO (276 m/s): 3 moves	1–3 (provides range of disposal options)
Payload Mass	GEO satellites: 500–2000 kg	500–2000 kg (reasonable range)
Payload Power	GEO satellites: 0.5–5 kW	0.5–5 kW (reasonable range)

3. TECHNOLOGY DESCRIPTION

Chapter 3 focuses on the technologies compared in the OECS. The chapter begins with a few thoughts on using the technologies and using hydrogen as a propellant. This is followed by a discussion of the individual technologies. Each discussion consists of an introduction and a system description. The introduction outlines the principles of operation of the technology, provides background, and summarizes results of previous research. The system description discusses the propulsion and power subsystems, including the components and the performance data. Appendix B contains additional information on the technologies and discusses the sizing relationships used to model the integration of these technologies into the upper stages.

Not all the innovative upper stages perform in the traditional sense of delivering the satellite and then separating. Only the baseline, advanced cryo, and solar thermal systems follow this pattern. Nuclear bimodal, nuclear electric, solar bimodal, and solar electric are integral upper stages, and they remain in whole or in part with the satellite throughout the satellite's operational life. Figure 3-1 illustrates the different upper stage concepts.

In addition to the separating/integral distinction, the technologies are differentiated by whether they are scalable or fixed. A scalable design can be customized to meet payload requirements. Baseline chemical and advanced cryo are fixed designs. Nuclear bimodal has a fixed reactor design, but its H_2 propellant tank and elements of its electrical power system are scalable. All other technologies are fully scalable.

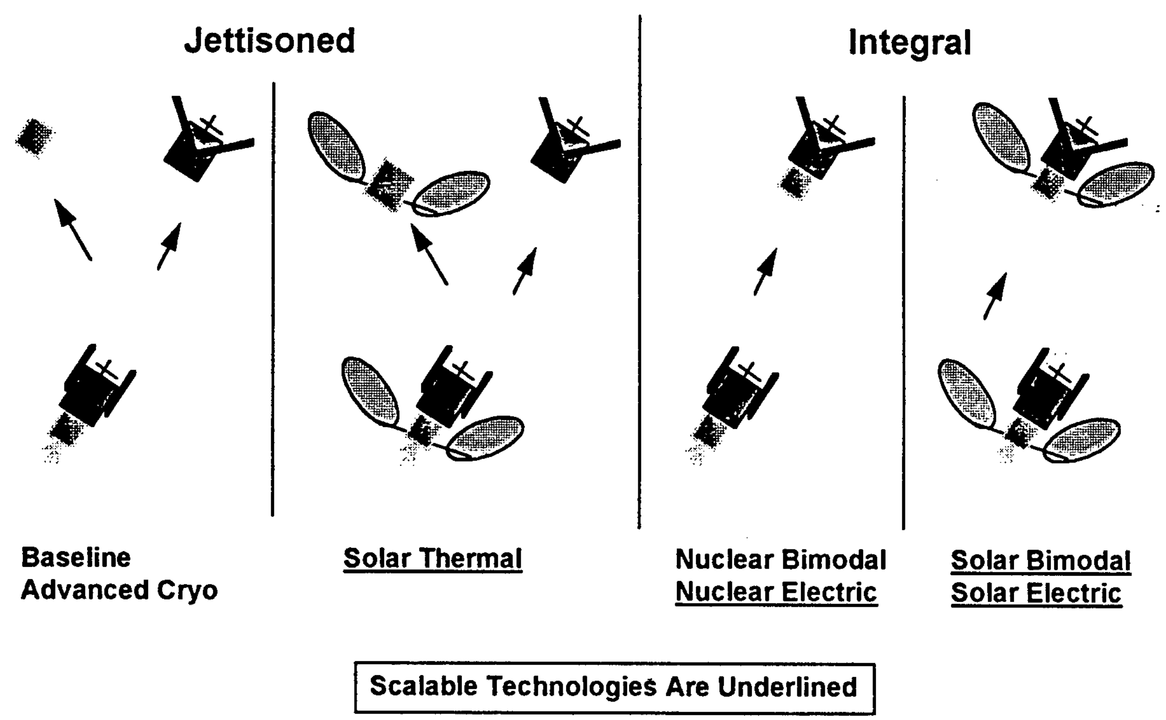


Figure 3-1. OECS upper stages.

Several of the innovative technologies use hydrogen as a propellant, all of them except advanced cryo without an oxidizer. However, using hydrogen as a propellant presents special challenges when integrating the system into existing launch vehicles. Liquid hydrogen has a very low density and consequently requires a correspondingly large propellant volume. As a result, in many instances there appears to be insufficient volume in the launch vehicle fairing to accommodate the upper stage and satellite (see Chapter 5, Figures 5-5, 5-6, and 5-7). Fairing modifications may be possible, but they raise a number of potential issues regarding launch vehicle limitations and the cost associated with modifying the launch vehicle and supporting facilities. Fairing modifications are beyond the OECS scope.

3.1 BASELINE CHEMICAL/DIRECT SYSTEMS

3.1.1 Introduction

The current fleet of launch vehicles and upper stages serves as the baseline for the OECS. The chemical upper stages considered in the OECS are listed by ORM and launch vehicle in Appendix B, Table B-3. ORM and launch vehicle determine whether an upper stage is needed. When an upper stage is required, the satellite separates from the upper stage after achieving the desired orbit. In cases where a launch vehicle with or without an upper stage does not place the satellite into its final orbit (e.g., the launch vehicle delivers the spacecraft to GTO), the satellite typically has a small, integrated, bipropellant propulsion subsystem to complete the orbital transfer.

3.1.2 System Description

3.1.2.1 POWER SUBSYSTEM

Power is provided by advanced, rigid, multijunction GaAs arrays with a 9.1-kg deployment mechanism. Array characteristics include 21% efficiency and 245.84 W/m^2. For HEO, a 30-mil frontal cover glass is assumed (resulting in a specific power of 34.87 W/kg); all other ORMs assume 4-mil cover glass (specific power of 47.59 W/kg). Energy storage is provided by nickel hydride (NiH) common pressure vessel batteries (49 W-hr/kg). The remaining power component is the power management and distribution system, which consists of regulators/converters and wiring harnesses.

The solar cells are sized on EOL power requirements. BOL power is EOL power divided by solar cell degradation, which is determined by the ORM and the satellite's design life.

3.1.2.2 PROPULSION SUBSYSTEM

An integral bipropellant subsystem provides post-upper stage transfer for those missions requiring an additional boost. On-orbit propulsion options are bipropellant chemical, monopropellant chemical, and hydrazine arcjets. The bipropellant systems use hydrazine (N_2H_4) and nitrogen tetroxide (N_2O_4) if the spacecraft has N_2H_4 already on board; otherwise, they use monomethyl hydrazine (MMH) and N_2O_4. The thrusters are off-the-shelf. The integral bipropellant transfer thrusters have an I_{sp} of 311 s; the bipropellant thrusters used for stationkeeping have an I_{sp} of 289 s. The monopropellant N_2H_4 thrusters used for minor stationkeeping and reaction control use a small off-the-shelf thruster with an I_{sp} of 225 s. The N_2H_4 arcjets are based on an off-the-shelf system and have an I_{sp} of 500 s.

3.2 ADVANCED CRYOGENIC

3.2.1 Introduction

Cryogenic propulsion systems, because of their higher I_{sp} than traditional storable bipropellant systems, have been used since the 1960s for high energy upper stages and for interplanetary missions. However, cryogenic hydrogen is expensive to store and handle, and it requires added tank structure weight. The advanced cryogenic upper stages for the OECS are based on the integral modular engine (IME) design studied by Rockwell. The advanced cryogenic upper stages replace the Centaur on both the Titan IV and Atlas IIAS. The new Titan stage is 3 ft longer than the current Centaur. For the Atlas IIAS, the advanced cryogenic stage was designed to maximize the launch vehicle's performance to GTO. With the Delta II, the advanced cryogenic stage replaces the existing Delta second stage and the PAM upper stage. It has a total propellant mass of 35,000 lb. Table B-5 in Appendix B defines the advanced cryogenic upper stage usage by ORM and launch vehicle.

3.2.2 System Description

3.2.2.1 POWER SUBSYSTEM

Spacecraft power is provided by advanced GaAs arrays, which were discussed in Section 3.1.2.1.

3.2.2.2 PROPULSION SUBSYSTEM

Rocketdyne's proposed integrated modular engine (IME) produces 45,000 lbf of thrust at a chamber pressure of 1195 psia. It has an expansion ratio of 160:1 and an I_{sp} of 467.5 s. Its length is 105 in and its estimated mass is 825 lbm. Its thrust-to-weight ratio is 55:1, which is about nominal for an expansion ratio of 160:1.

The advanced cryogenic stage for the Titan vehicle has a burnout mass of 2,082 kg (4,589 lbm). It has a useable propellant load of 20,412 kg (45,000 lbm), which results in a propellant mass fraction of 0.907. This is a significant improvement over the existing Titan Centaur (3,538 kg [7,800 lbm] burnout mass, 20,320 kg [44,800 lbm] useable propellants, and 0.852 propellant mass fraction). That stage, originally designed for use in the Space Shuttle, is considered by most sources to be heavier than necessary for Titan applications. The Atlas Centaur with the RL-10A-4 engine would be a better frame of reference for comparison with the IME. Its numbers are 2,177 kg (4,800 lbm) burnout mass, 16,783 kg (37,000 lbm) useable propellant, and a propellant mass fraction of 0.885. The postulated advanced cryogenic upper stage represents an increase of 2.2 % in propellant mass fraction. Table B-6 in Appendix B summarizes the advanced cryogenic stage for each launch vehicle.

3.3 NUCLEAR BIMODAL

3.3.1 Introduction

Figure 3-2 illustrates the nuclear bimodal system stowed and operational. A bimodal power and propulsion system combines in a single plant the ability to provide direct thermal propulsion and electric power. Nuclear bimodal spacecraft derive their power and primary propulsion from a single nuclear reactor. Propulsion is provided by expanding a gas (hydrogen or ammonia) by passing it through channels in the reactor. The system is also designed to convert reactor heat into electricity for the spacecraft. The concept of using a single space reactor to produce both direct thermal propulsion and spacecraft electrical power has been studied for over 20 years.

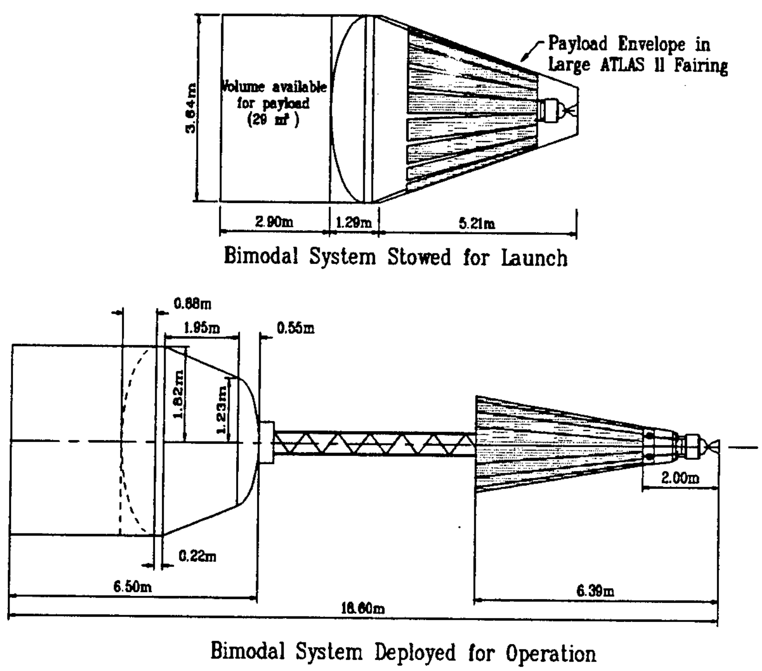


Figure 3-2. Nuclear bimodal system configured for ATLAS large payload fairing.

The principal elements of a nuclear bimodal system are the reactor core, radiation shield, power conversion, heat rejection system, and the propulsion system hardware. Liquid metal heat pipes transport energy to the power conversion system and distribute waste heat to the radiators. Figure D-1 in Appendix D shows how the reactor, H_2 tank, and payload are placed for launch.

Launch safety and disposal are issues of paramount importance to a nuclear bimodal system. The Russians have extensive experience with nuclear reactors in space with their Radar Ocean Reconnaissance Satellite (RORSAT). RORSAT has been operational for over 25 years. At the end of a RORSAT's life, the reactor separates and is boosted up to a 900 km circular orbit for the remainder of its 500 to 600 year life. The failure of a reactor to separate in 1978 provoked a serious international incident. Radioactive debris was scattered over an 800-km strip of land in Canada's Northwest Territories. The US also had an accident. In 1964 a Navy satellite with a nuclear generator failed to reach orbit and released radioactive material over the Indian Ocean. Neither accident resulted in the loss of life, but the events illustrate the problems inherent with placing and maintaining a reactor in space.

3.3.2 System Description

3.3.2.1 POWER SUBSYSTEM

The reactor and heat rejection system can be integrated with a variety of different power conversion technologies. As a result, the scalability of the power system is favorable over a wide range of electrical power levels. Figure 3-3 shows engine mass, excluding propellant tank and propellant, as a function of electric power production and power conversion technology (unicouple thermoelectric, multicouple thermoelectric, and alkali metal thermal-to-electric conversion [AMTEC]).

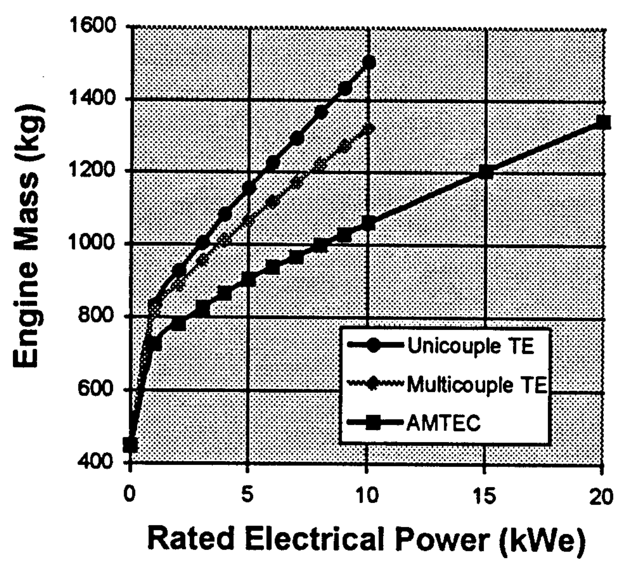


Figure 3-3. System mass vs. electrical power level.

For the OECS, multicouple thermoelectric diodes were selected for the power conversion system in part because of their past performance. The Voyager deep space mission, powered by a radioisotope thermoelectric generator (RTG), demonstrated the reliability and longevity of thermoelectrics for providing power in space. The SP-100 program offered significant performance improvements over previous thermoelectric power conversion technologies, such as RTG, and served as the performance baseline for multicouple thermoelectric diodes.

Deep space exploration requires an electric power source that is independent of the sun. The electrical power provided by the nuclear bimodal system could provide for active rather than passive sensors and increased data transmission rates. Nuclear bimodal would also appeal to commercial and military operators with high satellite electric power requirements, such as a space-based radar.

3.3.2.2 PROPULSION SUBSYSTEM

Transfer propulsion is provided by the nuclear bimodal system using hydrogen as the propellant. The OECS used a fixed nuclear bimodal point design: NEBA-1, Concept 3. The reactor core utilizes refractory metal cermet fuel elements developed and extensively tested in the 1960s. This fuel can accept prolonged exposure (hundreds of hours) to a variety of propellants including hydrogen and ammonia. The nuclear fuel serves as the heat source for a bleed cycle engine similar to many rocket engines currently in use today. NEBA-1, Concept 3 produces 2200 N thrust with an I_{sp} of 820 s. The hydrogen tank remains with the system upon reaching final orbit.

3.4 SOLAR BIMODAL

3.4.1 Introduction

Solar bimodal upper stages provide spacecraft power and propulsion. A solar bimodal system is an integral design which remains with the satellite after mission orbit is reached. The solar bimodal system unites aspects of solar thermal propulsion and solar thermal power systems that have been studied for over 30 years. Solar bimodal uses off-axis parabolic collectors to focus sunlight into one or two refractory metal receivers. The collected energy is used to heat a propellant, typically hydrogen, to high temperatures before it exits the system through one or more nozzles to produce thrust. In the electrical power mode, the same parabolic collectors focus sunlight into a high-temperature receiver where power conversion devices are located. A graphite thermal energy storage (TES) module is incorporated in the OECS design. In this configuration, thrusting is limited to short periods at perigee and apogee. The TES module is heated during the sunlight portion of an orbit, then heat is extracted during thrusting. Using the TES module to provide propulsive power allows a reduction in collector size. The TES module also provides continuous power production during eclipse periods in the final mission orbit.

The system consists of a pair of rigid collectors; support struts connected to a turntable; propellant tank; deployable boom; and a receiver and power conversion system positioned at the focal point of the collectors. A solar bimodal system is depicted in Figure 3-4. For the OECS, the payload is located on top of the propellant tank and the collectors are stowed in the annulus between the tank and the launch vehicle shroud (Appendix D, Figure D-3).

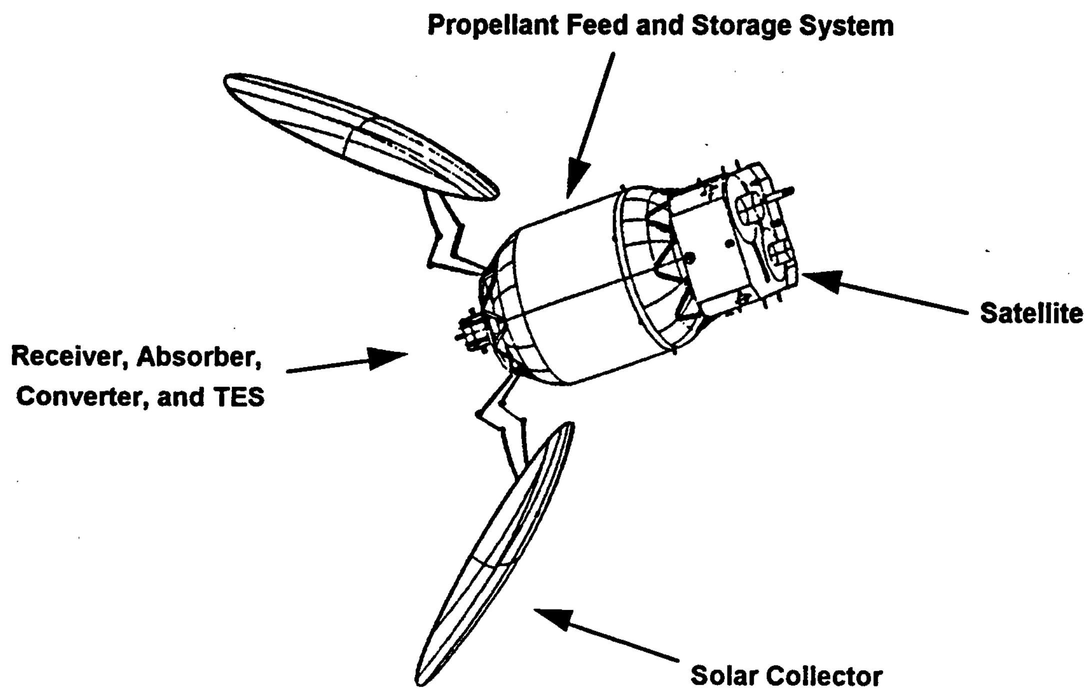


Figure 3-4. Solar bimodal system.

3.4.2 System Description

3.4.2.1 POWER SUBSYSTEM

Thermal power systems have been studied using Rankine, Brayton, Sterling, thermoelectric, thermionic, and AMTEC power-conversion devices. Some of the systems, such as the NASA solar Brayton power systems, utilize a TES module to allow continued power generation during eclipse periods. Figure 3-5 shows engine mass as a function of electrical power level for thermionic diodes, thermionic and AMTEC diodes, and inflatable collectors. As with Figure 3-3, engine mass excludes propellant tank and propellant.

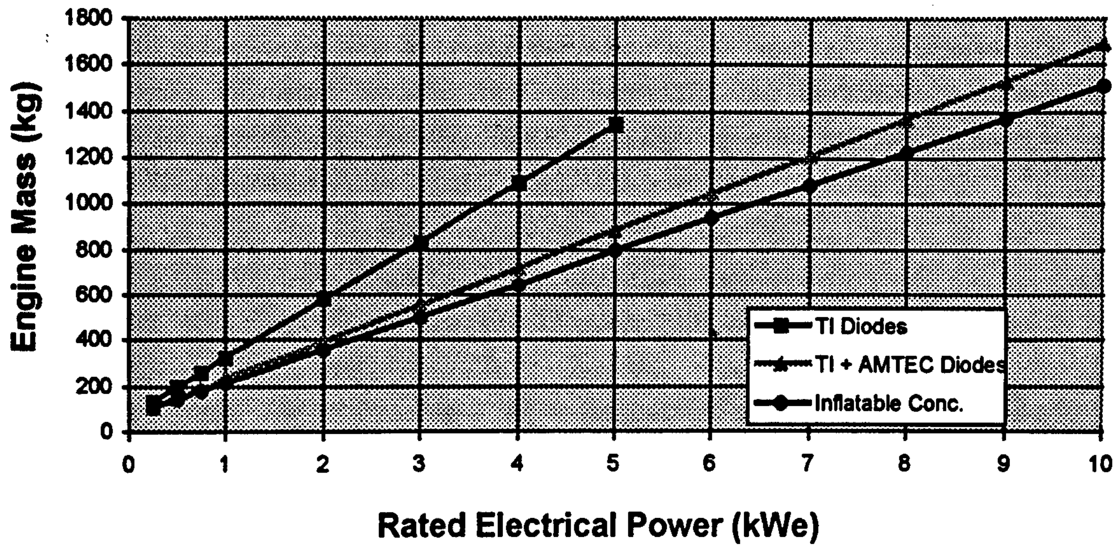


Figure 3-5. System mass vs. electrical power level.

The OECS evaluated a solar bimodal system that used thermionic power conversion and a TES module. Thermionic power conversion diodes are built into the receiver to allow the stage to generate electricity for the spacecraft. This eliminates the need for photovoltaic panels except for small body-mounted arrays to provide backup power for the spacecraft bus.

When not producing thrust, the energy entering the receiver from the collectors maintains the receiver structure near 2600 K. Energy is transferred radiatively to thermionic diodes located around the circumference of the receiver. The diodes operate with emitters between 1950 K and 2100 K to provide electrical power for the spacecraft. The system produces electricity with a net efficiency of approximately 10%.

3.4.2.2 PROPULSION SUBSYSTEM

The solar bimodal engine is a flexible system capable of providing either very low thrusts for continuous burn transfers (6 N for a 3-kWe system) to moderate thrusts for impulsive burns (approximately 50 N). At lower thrust levels, sunlight reflected by two parabolic collectors is used to heat the propellant to approximately 2500 K. Higher thrust levels require extracting heat from the TES module. During a higher thrust burn, propellant exhaust temperature varies from 2500–1500 K as heat is removed from the graphite. The solar bimodal engine consists of collectors, receiver, power conditioning system, and pointing and tracking system.

The two elliptical-shaped paraboloidal collectors provide approximately 0.83 kW/m^2. The receiver is a metal structure located at the focal point of the two collectors. The power conditioning system matches electrical output with instantaneous satellite power demands. The pointing and tracking system maintains correct collector pointing in two axes during all spacecraft maneuvers.

3.5 SOLAR THERMAL

3.5.1 Introduction

A solar thermal rocket propulsion design first appeared nearly 40 years ago. Solar thermal propulsion uses concentrated energy from sunlight to heat a working fluid. The working fluid, typically hydrogen, thermodynamically expands and accelerates out a nozzle, creating thrust. The system consists of a pair of large, inflatable, off-axis paraboloidal collectors, support struts connected to a turntable, a liquid hydrogen tank, and a heat exchanger/absorber positioned at the focal point of the collectors. Concepts have been envisioned with collectors as large as 30 m in diameter and with an I_{sp} exceeding 1,000 s achieved by using heated LH_2 as the propellant. A pair of collectors projected to be 30 m in diameter would intercept approximately 2 MW of sunlight. Figure 3-6 illustrates the conceptual, solar-powered upper stage rocket considered in this report.

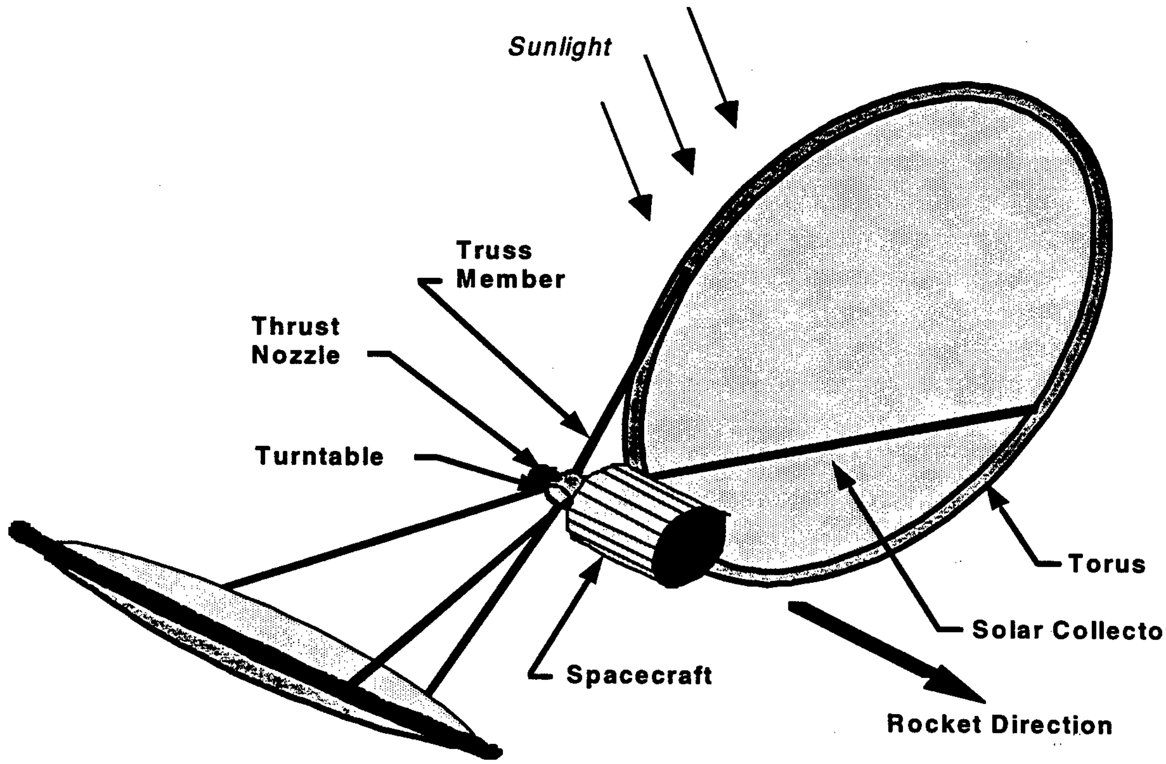


Figure 3-6. Solar thermal concept.

The OECS solar thermal propulsion technology is not an integral system: it delivers a spacecraft to a preliminary orbit and then separates. The preliminary orbit was assumed to occur in a typical EOL disposal orbit for safety reasons. After separation, the satellite's hold/move thrusters place the spacecraft in its operational orbit. For GEO, the separation orbit is 555 km (300 nmi) above GEO; for MEO-GPS, it is 93 km (50 nmi) above MEO.

The concentrator must accurately track the sun regardless of direction of travel; otherwise, concentrated sunlight will not reach the aperture and no thrust will be produced. Solar thermal does not have any thermal storage and will not produce thrust during an eclipse.

3.5.2 System Description

3.5.2.1 POWER SUBSYSTEM

Spacecraft power is provided by advanced GaAs arrays and is discussed in Section 3.1.2.1.

3.5.2.2 PROPULSION SUBSYSTEM

The mirror surfaces and opposing canopy membranes form gas-filled envelopes that are inflated once the upper stage reaches LEO. The envelopes are supported by mechanically deployed rigid tori. These tori keep the envelopes from expanding into a spherical shape. Mirror surfaces are made from reflectorized plastic sheets of NASA-Langley polyimide film molded in such a way the surfaces form sections of a parabola of revolution once the envelopes are inflated. The struts are also mechanically deployed rigid structures. Two axes of rotation allow the collectors to point toward the sun. A turntable that rotates 360 deg about the absorber entrance provides one degree of freedom, while rolling the entire rocket provides the other. The alignment of the collectors must be within a small fraction of the angular diameter of the sun, which is approximately 0.5 deg. Since the concept provides no thermal storage, the collectors are sized to support the maximum thrust.

A rigidized system would require added mass and deliver less payload to orbit. In addition, there is a greater likelihood that the inflatable concept achieves the desired reflector surface quality, especially for small reflectors. Inflatable mirrors are subject to eventual puncture by micrometeoroids. This limits the concept to orbital lift. If there were a puncture during lift, gas leakage would be very slow due to the low pressures, and short-term losses could be replaced.

Typical thrust values for a solar thermal system traveling to GEO are given in Table 3-1. These figures for Titan assume a 30-day trip time and a chamber temperature of 2778 K.

Table 3-1. Payload Delivered to ORM 1 (GEO) by Solar Thermal Lift as a Function of Incident Solar Power

Incident Solar Power (kW)	Thrust (N)	Payload (kg)
200	6.0	4,665
400	12.1	17,720
500	15.1	19,984
1,000	30.2	22,821
2,000	60.4	22,442

3.6 NUCLEAR ELECTRIC

3.6.1 Introduction

Electric propulsion has been tested over thirty years of space flight experience. Despite this extensive history, electric propulsion is not widely used by the military or commercial sectors. This lack of widespread acceptance can be attributed to several technical considerations, including thruster performance and lifetime, power source availability and mass, and electromagnetic interference. However, recent technology developments in high-power thrusters, autonomous guidance, navigation and control, and solar cell arrays can support an expanded role for electric propulsion.

A nuclear reactor provides power for the propulsion and remains with the spacecraft throughout its life. Nuclear electric propulsion provides thrust either by directly heating propellant gas and expanding it through a nozzle or by accelerating an ionized gas with an electric field.

Nuclear electric propulsion has many of the same associated subsystems as nuclear bimodal: reactor, radiation shielding, boom and structure, heat rejection, and reactor power conditioning and control. Outwardly, a nuclear electric system would look similar to a nuclear bimodal system (see Figure 3-2). Nuclear electric would have the same associated launch and disposal issues as nuclear bimodal (see Section 3.3.1).

The long GEO trip times associated with nuclear electric orbital transfer mean satellites will spend considerable time in the Van Allen belts. Possibly a slight increase in the satellite's hardness and shielding will be sufficient to address this issue. Long trip times also make storage of cryogenic hydrogen an issue. Additionally, the electromagnetic and plasma interaction with the spacecraft are not well understood. More work needs to be done.

3.6.2 System Description

3.6.2.1 POWER SUBSYSTEM

The nuclear power subsystem characteristics were based on historical information (SP-100, S-Prime, Topaz) analyzed by Rocketdyne (Appendix B, Figure B-2). The SP-100 program was sponsored by NASA, DOE, and BMDO and lasted from 1983 to 1994. The program sought to develop a 100-kW-class nuclear electric space system based on thermoelectric technology. Both the S-Prime and Topaz projects are thermionic-based power conversion technologies. S-Prime was a US effort that dates back to the 1960s. Topaz was an original Russia design the US bought in support of the nuclear-electric-propulsion space test program.

3.6.2.2 PROPULSION SUBSYSTEM

Although there are several categories of electric propulsion thrusters, the OECS evaluated three: arcjets, stationary plasma thrusters (SPT), and ion. These three technologies are within the ground rules for IOC and all can provide propulsion for lift, hold, and move. Arcjets are an electrothermal thruster with an I_{sp} of 300–1400 s. Hydrogen is the preferred propellant because of its high I_{sp} (1000–1400 s). A 1.8-kW hydrazine arcjet and power processor have been qualified for a 12-yr, north-south station keeping (NSSK) mission.

SPT and ion are both electrostatic thrusters, which produce thrust by ionizing noncontaminating inert gas with an electric field. SPT is a high efficiency Xe ion system (52%) with an I_{sp} of 1600–2000 s. The Russians have over 30 years of experience with SPT thrusters. The first system was launched in 1971. Since then over 50 different versions have flown.

Ion thrusters (I_{sp} 3000–8000 s) generate thrust by ionizing a low pressure working fluid and then accelerating it with a voltage grid. The beam is neutralized after acceleration so no charge builds up. Xenon is the optimal propellant, but it costs as much as ten times more than equivalent amounts of krypton and argon. Xenon is an expensive propellant, which is found in concentrations of approximately 90 parts per billion in air. Xenon availability should not be an issue, but the cost could be as high as $30 per standard liter (Wells). European-developed ion stationkeeping is scheduled for testing later this year.

Table 3-2 lists the I_{sp} and expected lifetime of the thruster/propellant combinations considered in the OECS. Where required, multiple sets of thrusters were included in OECS designs to circumvent lifetime limitations.

Table 3-2. Characteristics of the Options for Electric Propulsion Lift

Propellant & Thruster	I_{sp} (s)	Lifetime (hr)
Ammonia Arcjets	800	2,000
Hydrogen Arcjets	1200	2,000
Hydrazine Arcjets	550	2,000
Xenon SPT	1600	10,000
Xenon Ion	3200	10,000

3.7 SOLAR ELECTRIC

3.7.1 Introduction

The solar electric spacecraft is similar to the nuclear electric spacecraft except photovoltaic arrays provide electrical power instead of a nuclear reactor. Solar electric propulsion technology, like nuclear electric, is an integral system: it remains with the satellite in orbit (tanks and arcjets separate in the hydrogen arcjet case). As with solar thermal technology, the hydrogen arcjets and tanks separate from the spacecraft in disposal orbit. For GEO, the spacecraft is brought to an altitude 300 nmi above GEO, separation occurs, and the spacecraft uses its on-orbit propulsion system to return to GEO; for MEO, the separation altitude is 50 nmi above MEO.

3.7.2 System Description

3.7.2.1 POWER SUBSYSTEM

The power subsystem is based on advanced, flexible GaAs arrays that are 21% efficient and have a specific power of 245.84 W/m^2. They produce 61.1 W/kg. The arrays are a derivative of the Advanced Photovoltaic Solar Array (APSA). Radiation protection is the equivalent of approximately 12 mil of top cover glass. The bottom substrate provides the equivalent of approximately 12-mil cover glass, the same as our rigid arrays. The deployment mechanism consists of a canister and a boom.

Key to the sizing of the solar arrays are a set of curves derived from the solar array characteristics and output from the EVA Program. The program was used to characterize an "average" solar electric transfer system for a 300-day GEO transfer. The results vary slightly depending on the specific transfer technology. The curves (Appendix B, Figure B-1) show trip time/burn time, BOL thrust/EOL thrust, and EOL power/BOL power as a function of starting altitude.

3.7.2.2 PROPULSION SUBSYSTEM

The propulsion subsystem is identical to the nuclear electric propulsion subsystem described in Section 3.6.2.2.

4. ANALYSIS STRUCTURE AND METHODOLOGY

The goal of the OECS is to produce an unbiased comparison of the baseline and innovative technologies within study constraints. Here we summarize how we made these comparisons and present the underlying methodologies. We begin by discussing the major elements of the OECS within the framework of a generic analysis. Next we discuss the key analysis concepts in the study. This is followed by a summary of the critical methodologies, including details of how we selected the most applicable combinations of lift/hold/move/power technologies and how we chose to employ these technologies for the purposes of the study. The last section in the chapter describes the OECS Cost/Engineering Model (OCEM).

4.1 ANALYSIS OVERVIEW

The analysis community approaches a cost-effectiveness analysis by asking a standard set of questions. These questions have been institutionalized in the last few years in the cost and operational effectiveness analysis (COEA) process. The standard questions and the corresponding COEA elements are:

- *What is the job?* (mission needs statement [MNS], functional objectives [FOs], scenarios)
- *What are the alternatives for doing the job?* (alternative concepts)
- *How do I employ the alternatives?* (concept of operations [CONOPS])
- *How effective are the alternatives?* (measures of effectiveness [MOEs], measures of performance [MOPs])
- *What do the alternatives cost?* (cost analysis)
- *How cost effective are the alternatives?* (cost-effectiveness analysis)

The OECS asks and answers these questions for the innovative space propulsion and electrical power concepts. While there is no applicable propulsion or power MNS, the effectiveness analysis is based upon a hierarchy of FOs, MOEs, and MOPs.

4.1.1 What Is the Job? (Mission Needs Statement, Functional Objectives, Scenarios)

The job of OECS is to analyze the cost and effectiveness of combinations of technologies that provide propulsion to lift, hold, and move satellites and electrical power to operate the satellites. We use the terms *lift*, *hold*, *move*, and *power* to mean the following:

- *Lift* is defined in the OECS as "taking a satellite from a parking or transfer orbit and delivering it to its initial mission orbit." Depending upon the launch vehicle and initial mission orbit, lift today is done primarily with expendable chemical upper stages and separate (or integrated) apogee kick motors.

- *Hold* is defined as "providing satellite stationkeeping to maintain a satellite's orbital elements within tolerance." Stationkeeping has traditionally been done with cold gas or hydrazine (N_2H_4) monopropellant thrusters. However, bipropellants can be used, and hydrazine arcjets are state of the art for stationkeeping.

- *Move* means "changing from a storage orbit to an operational orbit, or from an operational orbit to another operational orbit, or to a disposal orbit at the satellite's end of life." Because a move typically requires significant expenditures of onboard propellant, a move from one operational orbit to another has been *ad hoc* rather than a part of standard operating procedures.

- *Power* means "supplying a satellite with housekeeping and payload electrical power throughout its life."

These tasks are examined in the OECS in a variety of scenarios corresponding to the ORMs introduced in Chapter 2. Because there is considerable latitude in each ORM's constellation size and satellite parameters (payload mass and power, MMD, etc.), we are able to compare technology performances over a range of situations.

4.1.2 What Are the Alternatives for Doing the Job? (Alternative Concepts)

Every operations analysis needs a baseline against which other options are measured. The baseline launch vehicles, with the baseline upper stages in parentheses, are:

- Delta II (PAM DII upper stage)
- Atlas IIAS (Centaur upper stage)
- Titan IV (SRMU, no upper stage [NUS])
- Titan IV (SRMU, Centaur upper stage)

Comparisons are made by replacing or augmenting the baseline upper stages with the innovative upper stages. The Lockheed Launch Vehicle 3 (LLV3) serves as an additional launch vehicle for some of the innovative upper stages to determine launch vehicle stepdown.

The baseline propulsion for on-orbit satellite stationkeeping and maneuver consists, as appropriate, of hydrazine (N_2H_4) thrusters and hydrazine arcjet thrusters. Baseline electric power is not supplied by existing photovoltaic cells; rather it is supplied by advanced photovoltaic solar array (APSA) cells, which are available to all technologies (see the photovoltaics ground rule in Section 1.6).

The innovative technology options considered in the OECS were introduced in Chapter 1 and discussed in more detail in Chapter 3. Briefly, they are:

- Advanced cryo (advanced cryogenic propulsion and photovoltaic power)
- Nuclear bimodal (nuclear thermal propulsion and thermoelectric power)
- Solar bimodal (solar thermal propulsion and thermionic power)
- Solar thermal (solar thermal propulsion and photovoltaic power)

- Nuclear electric (nuclear electric propulsion and thermionic power)
- Solar electric (solar electric propulsion and photovoltaic power)

These innovative propulsion and electrical power technologies were selected because they offer potential near-term options that may be more effective or cost effective than the baseline technologies. A ground rule developed during the study requires a reasonable expectation that, given an adequate development program, the innovative technologies could have a flight demonstration within seven years and an initial operational capability (IOC) within ten years.

In addition to the major propulsion technologies, there are five electric propulsion subtechnologies: ammonia (NH_3), hydrazine (N_2H_4), and hydrogen (H_2) arcjets, xenon SPT thrusters, and xenon ion propulsion.

Within a wide latitude, these technologies can be combined to perform the lift, hold, move, and electric power tasks for each ORM. The job of selecting the technology and subtechnology combinations that make sense is critical to the study. The selection methodology and its results are discussed in detail in Section 4.3.8. The selection process is based on 15 ground rules covering propulsion and electrical power technologies.

The potential for the greater cost effectiveness of the innovative technologies resides in their high specific impulses (I_{sp}), a measure of the total impulse achieved from a unit mass of propellant. The potential is tempered by future development costs; by potentially large physical structures and masses necessary in many cases to achieve high the I_{sp}'s; and, except for the advanced cryogenic technology, by moderate or very low thrust levels. Table 4-1 shows the approximate I_{sp} and thrust levels we used for each technology.

Table 4-1. Approximate Levels of Upper Stage I_{SP} and Thrust in the OECS

		Upper Stage Thrust (N)		
Technology	**I_{sp} (s)**	**Delta II**	**Atlas IIAS**	**Titan**
Baseline	Delta 292.6 Atlas 448.9 Titan 444	66,440	185,000	2 × 73,000
Advanced Cryo	467.5	200,160	200,160	200,160
Nuclear Bimodal	820	2200	2200	2200
Solar Bimodal	750–780	115–270	250–600	400–535
Solar Thermal	840–875	10–30	15–45	30–90
Nuclear Electric	H_2 Arcjet 1200[a] SPT 1600 Ion 3200	N/A	1.5–5	2–6
Solar Electric	H_2 Arcjet 1200[a] SPT 1600 Ion 3200	1–2	1–2	4–6

[a]N_2H_4 arcjets have a 550 s lift I_{sp}, NH_3 800 s.

Higher I_{sp} values often provide a given ΔV with less total propulsion mass (hardware plus propellant), increasing available on-orbit payload mass, payload power, or ΔV. On the other hand, lower thrust means longer times to apply a given ΔV. For electric propulsion, this can lead to lift times of many months to GEO and MEO. Thus, perhaps ironically, the greater the ΔV required, the greater the mass saved, and the greater the time required. These latter two factors represent the principal asset and principal shortcoming of all the low-thrust innovative technologies.

4.1.3 How Do I Employ the Alternatives? (Concept of Operations)

The CONOPS for the innovative technologies did not exist prior to the OECS. Fortunately the lift, hold, move, and electric power tasks can be defined from historical and dynamic considerations for each ORM. This allows basic CONOPS to be formulated. Many of these issues were discussed in Chapter 2. It remains in this chapter to interpret them with respect to each of the technologies. Section 4.3.9 discusses the best use of each technology for each ORM. Employment choices were made solely on technical grounds.

The large range of thrust characterizing the technologies provides a CONOPS challenge, especially for the lift function. Desirable upper stage drop-off conditions (altitude and orbital parameters), lift strategy (number and location of burns), and trip time will vary substantially with thrust level and, hence, with the technology. We have been able to group the technologies by thrust level to facilitate discussion of their employment. The three groups and their distinguishing propulsion characteristics are:

- High thrust: baseline chemical and advanced cryogenic (impulsive thrust)
- Moderate thrust: nuclear bimodal, solar bimodal, and solar thermal (multi-thrust)
- Low thrust: nuclear electric and solar electric (continuous thrust)

4.1.4 How Effective Are the Alternatives? (Measures of Effectiveness, Measures of Performance)

The effectiveness analysis focuses on four FOs:

FO 1: Lift satellites to initial operational orbits
FO 2: Hold satellites in operational orbits (i.e., provide stationkeeping)
FO 3: Move satellites to other orbits (i.e., maneuver)
FO 4: Power satellite payload and housekeeping operations

The MOEs are derived from the FOs and are used to gauge how well the FOs are met. The MOPs are developed to help evaluate the MOEs.

Quantitative and qualitative MOEs and MOPs have been developed to identify the best technology combinations if cost is not a consideration. Several quantitative MOEs/MOPs are based on lift performance. From the bottom-up perspective, a principal measure is how much payload mass a given launch vehicle and a lift/hold/move/power technology combination can place in operational orbit. From a top-down perspective, a principal measure is how much of a given payload can be placed on a smaller launch vehicle (step-down) for various lift/hold/move/power combinations. Both bottom-up and top-down methodologies are discussed as key analysis concepts later in this chapter.

Another quantitative MOE examines how varying the lift technologies affects constellation availabilities. Availability—the probability of having a fully operational satellite constellation—is a significant factor in the OECS. Availability is described in detail in Section 4.3.1. A complete discussion of all MOEs and MOPs is found in the next chapter.

4.1.5 What Do the Alternatives Cost? (Cost Analysis)

In addition to assessing the effectiveness of each design, the OECS estimates their cost. We determine an "acquisition cost" for development, procurement, and launch, and any unique facilities or launch vehicle modifications. We do not provide the life cycle costs found in COEAs because we have not estimated operations and support (O&S) costs for the satellite payloads or for the upper stage and satellite propulsion and power subsystems. Payload O&S costs are payload specific, and they can be large and difficult to determine. They are nominally independent of the propulsion and electric power technologies. Propulsion and power O&S costs are small, and they, too, are not likely to vary significantly among technologies. The addition of O&S costs, especially payload costs, would complicate the cost-estimating process and the comparison of results.

The cost work breakdown structure (WBS) includes:

- Technology acquisition cost required to mature each technology to the point of system development
- Flight demonstration cost, including launch cost and production cost of a demonstration unit
- Engineering and manufacturing development cost (formerly full-scale development cost)
- Unit production cost, including production learning
- Satellite constellation launch costs (used to highlight savings from launching the same payload on a smaller launch vehicle or from launching fewer satellites to maintain the same constellation)
- Representative payload acquisition cost (used to highlight cost savings from launching fewer payloads to maintain the same constellation)
- Unique and unavailable facilities cost
- Launch vehicle modifications cost, such as a new payload fairing

We determine the most likely total cost of the satellite constellation in each ORM by summing the most likely individual costs of each cost category. In a small sampling of cases, we perform a cost risk analysis by analyzing the impact of potential high and low component costs on the total cost. The risk analysis reports the 70th percentile as a high cost estimate and the 30th percentile as a low cost estimate of the total cost probability density function.

Cost ground rules, the detailed cost breakdown structure, and a summary of cost results are presented in Chapter 6. The cost estimating relationships (CERs) are documented in a separate electronic volume, Appendix F.

4.1.6 How Cost-Effective Are the Alternatives? (Cost-Effectiveness Analysis)

The cost-effectiveness analysis is presented in Chapter 7. We have chosen to do our cost-effectiveness analysis by looking at the cost to produce equal effectiveness. Our investigation lends itself to defining effectiveness in terms of the performance of a specific job: maintaining a constellation of operational satellites whose payload mass, payload electrical power, and hold and move capabilities are specified.

The independent variables of interest to the OECS cost and cost-effectiveness analyses are:

- Payload mass
- Payload electrical power
- Satellite mean mission duration (MMD)
- Number of "standard" satellite moves on orbit (ΔV)
- Number of satellites purchased

If we denote the functional relationship for cost as f, we can write

$$Cost = f(x_1, x_2, x_3, x_4, x_5, LV, tech, ORM)$$

where each x_i represents one of the independent variables in the above order, *LV* identifies the launch vehicle, *tech* specifies the lift/hold/move/power technology combination, and *ORM* is the mission orbit. The launch vehicle cost is a dependent variable determined by the first four independent variables, x_1 to x_4.

Attempts were made to approximate f using multiple linear regression (MLR). MLR creates multidimensional mathematical functions that approximate, in a least squares sense, a dependent variable in terms of multiple independent variables. Using MLR, f is found by fitting a response surface to a set of specially chosen points representing an experimental design. Since the resulting function is continuous, we could estimate the cost of any point on our five-dimensional cost surface within the ranges of the factors used to generate f, subject to errors of fit (approximation). This is an extremely powerful tool, for it permits wide-ranging cost comparisons of different technologies for relatively few cost determinations. Unfortunately, our MLR approximations had unsatisfactory accuracy, and we retreated to the less elegant and less flexible but more familiar parametric variations.

4.2 KEY ANALYSIS CONCEPTS

The previous section outlined the OECS analysis. This section describes key aspects of the analysis needed to place the study and its results in context. We begin with a discussion of two operational concepts: constellation availability and the response time needed to place an operational satellite on-orbit. Both are critical issues for several innovative technologies that have long response times compared to the alternatives. Next we examine the distinction between satellite mass and satellite payload mass . We show that payload mass is the proper analysis measure for the OECS because of the interactions of the innovative technologies with the satellites. This is followed by a discussion of scalable technologies and their use in sizing both upper stages and satellites. Many of these concepts come together in a discussion of bottom-up and top-down analysis. Finally, we conclude with the assertion that now is the time to be examining the potential impact of innovative technologies on future launch systems.

4.2.1 Constellation Availability vs. Response Time

Satellites are either suppliers or relayers of information. Since communications, location, weather, and early warning information is critical, we go to great lengths to provide it on a continuing basis by maintaining an adequate number of operational satellites on orbit. We want these satellites to have a high probability of functioning, i.e., we want the satellites and the constellation to be available.

While a number of factors affect constellation availability, the principal factor is the time needed to respond to an unforeseen satellite failure. The *Solar Electric Propulsion Assessment* (Chan et al.) demonstrated that a long response time is not unthinkable in combination with on-orbit spare satellites, and it may be cost effective if it allows less costly launches. This issue is discussed in more detail in Chapter 5. A possible alternative to having a full complement of satellites available on-orbit is quick augmentation of on-orbit satellites with new launches when needed. Unfortunately, a quick response time is projected to be very expensive (Schulenburg et al., pp. 5-2 to 5-17), and an augmentation strategy may not be affordable or even practical.

As a result of these considerations, the OECS analysis has focused strictly on using the innovative technologies to maintain a constant on-orbit capability.

4.2.2 Satellite vs. Payload Mass

Traditionally, when we think about launch vehicles and upper stages, we focus on the total mass a system can deliver to an orbit. This has been adequate in the past, for satellites have almost uniformly relied on photovoltaic cells for electrical power and hydrazine (N_2H_4) for on-orbit propulsion. The result, at least among functionally similar payloads, has been a more or less constant ratio between satellite mass and payload mass. This paradigm is inadequate for the OECS because the variety of innovative technologies has created many relationships between satellite mass and payload mass.

Functionally equivalent OECS satellite masses will vary considerably with both the propulsion and electric power technologies. Many of the innovative technologies will result in more or less massive satellites on-orbit for a given payload compared to the baseline. For example, the mass of a satellite that draws its electrical power from a nuclear reactor is apt to be substantially different from one employing photovoltaic cells. This diversity means we must use a new paradigm based on equal payload masses. This concept was introduced in Section 1.5 of Chapter 1.

4.2.3 Upper Stage and Satellite Sizing

The OECS does not require detailed upper stage and satellite designs (for example, exact placement or orientation of all components). It does require consistent and reliable estimates of their mass and, to a lesser extent, their dimensions. We have called this limited design process *sizing* to draw attention to its lack of fine detail. Sizing is dependent upon engineering scaling algorithms and databases that interrelate the sizes and masses of components. Sizing is accomplished with the OECS Cost/Engineering Model (OCEM), which is discussed in more detail in Section 4.4. Depending upon the component, sizing may be based on a continuous function (e.g., propellant tank volume) or on a selection of one of several discrete choices (e.g., arcjet thrusters).

The baseline booster and upper stage propulsion technologies are not scalable. Of the innovative technologies, only the advanced cryogenic technology and the nuclear bimodal reactor technology were considered not scalable. (The nuclear bimodal propellant tank and electrical components are scalable.) For the other innovative technologies, upper stage designs are scaled with OCEM. This is done, for example, by varying the size of the solar collector array and those aspects of the system that depend upon it, e.g., structure, plumbing and wiring, and thermal control.

Upper Stage Scalability

In the past, fine tuning of a launch vehicle's capability has generally meant improving or changing rocket engines, varying the number or type of strap-ons, using improved materials/fabrication processes, and exchanging or improving the upper stage. An ability to scale the upper stage more or less continuously may reduce the reliance on these traditional means and lower the cost in the bargain.

Of course, the cost effectiveness and practicality of scaling innovative upper stages to match the payload and ORM needs to be demonstrated. The solar technologies (bimodal, thermal, and electric) may offer the best prospects for scaling.

4.2.4 Bottom-Up and Top-Down Effectiveness Analysis

Depending upon the issue, we approach effectiveness from two perspectives: bottom up and top down. The bottom-up approach is used to determine the maximum satellite payload that can be placed in orbit by a given combination of propulsion and power technologies for a given launch vehicle. In this approach, the analyst begins with the launch vehicle, the ORM, and the move, hold, and electrical power requirements for the satellite, then sizes the upper stage and the satellite. In the top-down approach, the analyst begins with a defined payload on orbit (mass, electrical power, ΔV). He or she then designs the satellite and the upper stage needed to put it there for a specific technology combination and determines what launch vehicle is required to launch it. The bottom-up approach is the primary methodology for the effectiveness analysis; the top-down approach is used in the cost and the cost-effectiveness analysis.

Figure 4-1 shows the bottom-up and top-down inputs to the OCEM (shaded boxes) and the outputs from it (unshaded boxes). The concept sizing box (lighter shading) has the outputs used to determine the smallest usable launch vehicle, an input to the cost determinations. OCEM necessarily involves many interative loops because of the interrelationships among the subsystems. The complexity of OCEM is shown in Figure 4-6, which schematically illustrates the subsystem interactions.

4.3 CRITICAL METHODOLOGY

4.3.1 Constellation Availability

Cost-effectiveness comparisons are best performed by comparing effectiveness given equal cost or, more usually, by comparing cost given equal effectiveness. OECS follows the latter path. In this analysis, *equal effectiveness* means "equal probabilities of maintaining a specified number of essentially identical operational satellites (a constellation of satellites) on orbit for a specified period of time." The probability of interest is *constellation availability* (P_a), which is "the probability that the constellation is performing its required functions at any randomly chosen time after it has been established." The principal parameters determining availability are:

- Number of satellites in the constellation
- Number of on-orbit spare satellites
- Launch vehicle reliability
- Satellite MMD (reliability)
- Time to deploy a satellite
- Minimum time between successive satellite launches

An average number of satellites, upper stages, and launch vehicles need to be purchased to maintain a specified availability for some period. That number will vary with any of these parameters. As a result, the cost of achieving the specified availability will also vary.

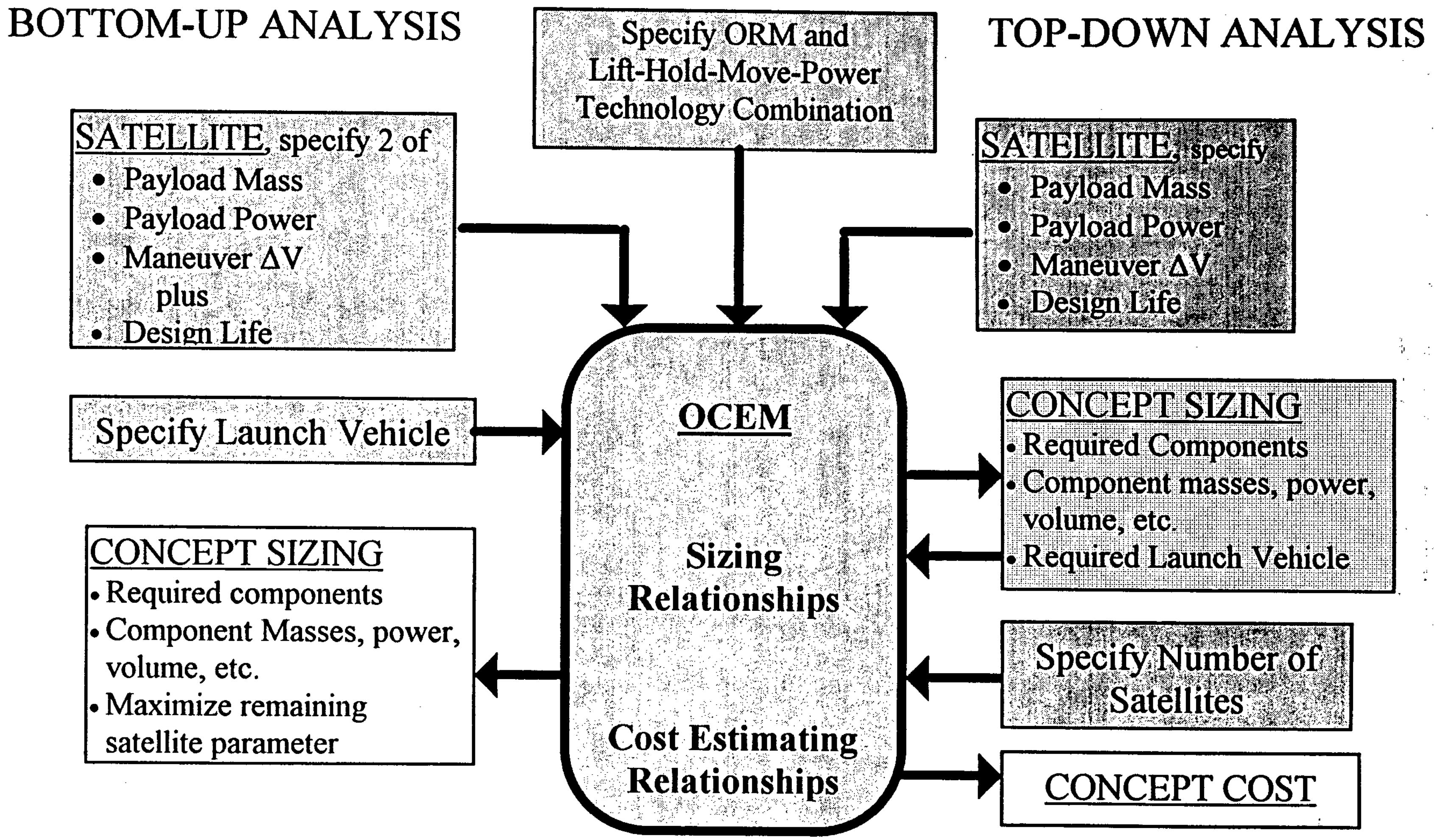


Figure 4-1. Inputs and outputs for the OCEM bottom-up and top-down analysis.

Since availability is complex to estimate, it is determined through Monte Carlo simulation of launch failures and operating satellite failures. This was done in the OECS with the GAP_PLUS model. The GAP_PLUS methodology is based substantially on the Aerospace GAP (Generalized Availability Program) model methodology. GAP_PLUS was written by the Office of Aerospace Studies and has been used in support of the *Comprehensive On-Orbit Maintenance Assessment (COMA)* (Feuchter et al.). The current version of GAP_PLUS has been updated for the OECS. A recent addition to the model allows specification of the minimum time between successive satellite launches.

When all factors are considered, typical maximum achievable availabilities lie in the range of 90 to 100% ($P_a = 0.9$ to 1.0). An availability of 100% is never achievable, and on-orbit spares are typically needed to support availabilities much in excess of 90%.

4.3.2 Constellation Size and On-Orbit Spares

A constellation consists of a specified number of operational satellites, which may include operational spares. The total number of operational satellites defines the constellation's size. Depending upon circumstances, the constellation may also contain one or more non-operational (dormant) spare satellites held in reserve as replacements for failed operational satellites. We only consider operational spares in the OECS. Constellations typically have been no larger than five satellites, with the notable exception of the 24-satellite GPS constellation with its 3 spares (21 + 3 satellites on-orbit). However, large constellations of smaller satellites are likely in the future.

4.3.3 Launch Vehicle Reliability

In the OECS, the term *launch vehicle reliability* refers to "the probability a satellite is launched and placed in its mission orbit." It is composed of the reliabilities of the booster and any transfer stages. Typical launch vehicle reliabilities are near 90% (Adams et al., pp. 3-24 to 3-29). A ground rule of the OECS effectiveness and cost-effectiveness analyses is that all launch vehicle reliabilities—and by implication all transfer stage reliabilities—are equal. This is an adequate approximation and eliminates the controversy of trying to assess reliabilities for a number of innovative technologies that have not flown. OECS did, however, attempt to qualitatively identify factors that may decrease or increase the reliability of each technology.

4.3.4 Satellite Reliability

Modeling satellite reliability in the OECS involves simulating satellite failures that require the satellite be replaced (some failures can be worked around to prevent the need for replacement). Two types of typical failures are modeled. Both are random failures, but each is described with a different probability distribution. One type of failure can occur at any point during the satellite's life. For lack of a better term, we will refer to these failures as *random failures*. The second type tends to be localized in time and relates to the design life (i.e., the maximum life) of the satellite. We will refer to these failures as *design life failures* (or truncation failures). Design life is an appropriate term because this category of failure is inherent in the satellite's design. Design life represents the predicted failure time of the satellite due to mechanical wearout, battery failure, photovoltaic cell deterioration, the exhaustion of consumable working fluids such coolant, etc. (As discussed in Section 2.1.2 of Chapter 2, *design life* is alternately used to refer to "minimum projected life.")

Figure 4-2 shows a typical satellite reliability plot for each type of failure. Overall satellite reliability is the product of the two reliabilities. Historically, satellite random failures have been modeled with the two-parameter Weibull distribution. One parameter, the α or scale parameter, is primarily related to satellite mean life. The β or shape parameter is related to the shape of the distribution. Weibull reliability is given by

$$R(t) = e^{-\left(\frac{t}{\alpha}\right)^{\beta}}$$

where t is the elapsed time. A representative β value for OECS-type analyses is 1.6.

Considering only R(t), the average satellite life or MMD is

$$MMD = \int_0^{\infty} R(t)dt = \int_0^{\infty} e^{-\left(\frac{t}{\alpha}\right)^{\beta}} dt$$

We also use the Weibull distribution to model design life failures. As shown in Figure 4-2, this distribution approaches a step function when we select a very high value for β (say 250). Using τ to indicate the mean of this distribution gives

$$R_{designlife}(t) = e^{-\left(\frac{t}{\tau}\right)^{\beta}}$$

Because of the step-like nature of the function, τ is essentially the mean design life. Assuming design life is a step function at $t = \tau$ allows us to approximate the combined satellite MMD as (Nishime, pp. 28–30)

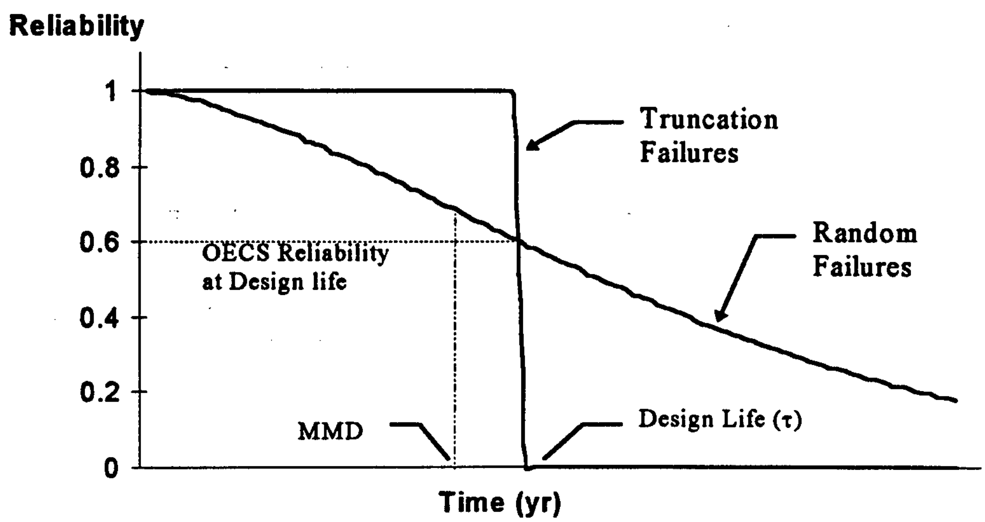


Figure 4-2. Typical satellite reliability.

$$MMD = \int_0^{\tau} R(t)dt = \int_0^{\tau} e^{-\left(\frac{t}{\alpha}\right)^{\beta}} dt$$

We have assumed in the OECS that $R(\tau) = 0.6$, i.e., the satellite reliability at design life is 0.6. Given this assumption, it can be shown (see Appendix A) that

$$\tau = 0.6572\alpha$$

and the satellite's average life or mean mission duration (MMD) is related to τ by

$$MMD = 0.8311\tau$$

4.3.5 Deployment Time

Deployment time means "the total time from the decision to launch a satellite to having it operational on orbit." Activities could include transportation of satellite and launch vehicle to the launch site, integration of the satellite and launch vehicle, prelaunch pad activities, launch, trip time (travel) to mission orbit, and satellite check-out. Today these activities can take many months, even though the trip time is, at most, only a few days.

Shortening the deployment time can reduce or eliminate the number of spare satellites needed on orbit to support a constellation. Conversely, lengthening the deployment time can increase the number of spares that are needed. All other things equal, fewer spares correspond to fewer launch vehicles and satellites—and, therefore, less money—needed to support a constellation over its lifetime. Short deployment times represent an improvement in ability to react to unforeseen events. However, even if we assume launch vehicle step-down is possible with an innovative technology, the *Solar Electric Propulsion Assessment* has shown that short deployment times are not necessarily the least expensive way to support a constellation at a given availability (Chan et al., pp. 5–62 to 5–88). This is an important consideration in the OECS cost-effectiveness analysis. Trip times for the electric propulsion technologies can be a significant fraction of a year, and thermal technologies can have trip times of one or two months.

4.3.6 Minimum Time Between Satellite Launches

The minimum time between launches was not considered in the *Solar Electric Propulsion Assessment* except for establishing the constellation. Typically this parameter has been assigned a value of zero; i.e., consecutive launches can be as close together as necessary. Obviously this is not realistic given limited launch pads and nonzero launch pad turnaround. Availability can be strongly dependent on minimum time between launches, especially for larger constellations, which have more failures per unit time. In the OECS, the same minimum interval was used between the launches to establish the constellation and the launches to maintain it (typically 2–4 months).

4.3.7 Other Considerations

Table 4-2 identifies other availability considerations in the model that are not directly related to the six parameters just discussed, and the assumptions we made about them.

4.3.8 Technology Selection

For each case examined by OCEM, we must specify

- ORM
- Lift technology
- Hold technology
- Move technology
- Electric power technology

Table 4-2. Other Modeling Choices Affecting Availability Determinations

Modeling Factor	Assumption
Constellation maintenance	All constellations maintained 15 years after establishment phase
Infant mortality (early satellite failure due to initial design flaws)	None
End of design life replacement	Replacement satellite scheduled on orbit 0.2 years before anticipated design life failure (for purposes of planning satellite/launch vehicle acquisition)
Satellite block changes	None
Constellation geometry	No grouping of satellites to account for multi-orbital plane constellations. Spare satellites, when present, are universal spares (can substitute for any failed satellite). GAP_PLUS test runs modeling GPS as six planes of four satellites each show no significant differences in availability or average number of satellites required over the single group approach.
Number of simulation replications per simulation	500

If we were to analyze each of the seven major categories of propulsion technologies that can be used to perform lift, hold, and move functions, we would get $7^3 = 343$ combinations of propulsion technologies per ORM. This number would grow substantially if we were to add in the specific technologies within the seven general categories. The number would grow even larger when the electric power options were added. Fortunately, many combinations can be eliminated. Many technology combinations are incompatible, some combinations are obviously more costly than others, and some combinations introduce unnecessary complexity.

A set of fifteen ground rules was developed to reject combinations not suited to the task at hand (Table 4-3). The first five rules apply to all ORMs. Rules 6 through 9 are ORM-specific, and 10 through 15 apply to electric power. In accordance with study philosophy, none of the ground rules are based on safety and policy considerations.

The ground rules were applied to each possible propulsion combination for each ORM. Combinations which violated one or more ground rules were eliminated. The results of this screening are shown in matrix format by ORM in Table 4-4 through

Table 4-7. Each six-by-six matrix represents a different lift technology. Rows represent move technologies; columns represent hold technologies. In this process, solar bimodal lift has been combined with solar thermal lift because of their similarity. Advanced cryogenic propulsion is a chemical technology.

In a matrix cell, numbers refer to the ground rules violated by the combination. Shaded cells indicate combinations that do not violate any ground rules. These cells contain abbreviations indicating the electric power choice(s): PV, photovoltaic; NB, nuclear bimodal; SB, solar bimodal; NE, nuclear electric. From the original possibilities, only 60 technology combinations were selected. These are listed by ORM in Table 4-8. While Table 4-4 through Table 4-7 do not contain propulsion subtechnologies, the selected subtechnologies are shown in Table 4-8. A summary of the subtechnologies appears in Table 4-9. The ground rules for selecting subtechnologies are given in Table 4-10, which lists state-of-the-art propulsion solutions and selected innovative propulsion solutions as functions of hold and move ΔVs.

Table 4-3. Ground Rules and Rationale Used to Select Combinations of Technologies Evaluated in the OECS

No.	GROUND RULE	RATIONALE
	ALL ORMs	
1	No CRYO MOVE or CRYO HOLD	Difficulty of storing cryo propellant for a long duration coupled with complexity and mass considerations
2	No mixing of NUCLEAR and SOLAR systems	Cost, complexity, and mass considerations
3	NUCLEAR MOVE and NUCLEAR HOLD can only be used with NUCLEAR LIFT	Cost, complexity, and mass considerations
4	THERMAL MOVE can only be used with THERMAL LIFT	Cost, complexity, and mass considerations
5	THERMAL HOLD not allowed	Requires multiple hot gas valves
	SPECIFIC ORMs	
6	No CHEMICAL HOLD for ORM 1	Hydrazine arcjet is state of the art for ORM 1 N-S stationkeeping (too high a ΔV to justify chemical hold)
7	Only CHEMICAL MOVE and HOLD are used for ORMs 2 and 3a,b	Move and hold ΔVs are too low to justify other choices
8	Only CHEMICAL LIFT, CRYO LIFT, and SOLAR ELECTRIC LIFT apply to ORM 3a,b	Lift ΔV too low to justify other choices
9	HOLD and MOVE technologies identical for ORMs 2 and 3a,b	Move not an option or move performed with hold system
	ELECTRIC POWER GENERATION	
10	CHEMICAL LIFT is always combined with photovoltaic electric power	Minimally massive and minimally complex electric power system
11	CRYO LIFT is always combined with photovoltaic electric power	Minimally massive and minimally complex electric power system
12	NUCLEAR THERMAL LIFT is always combined with nuclear bimodal electric power	Electric power capabilities of nuclear bimodal system will meet satellite electric power requirements (design ground rule)
13	SOLAR THERMAL LIFT is always combined with either solar bimodal or photovoltaic electric power. For pure (non-bimodal) solar thermal lift, the solar thermal system is discarded upon reaching mission orbit because the collectors have a short design life.	Electric power capabilities of solar bimodal system will meet satellite electric power requirements. However, potential technical and operational difficulties make photovoltaics a potential viable alternative.
14	NUCLEAR ELECTRIC LIFT is always combined with nuclear electric power	No other sensible choice
15	SOLAR ELECTRIC LIFT is always combined with photovoltaic electric power	No other sensible choice

Table 4-4. Matrix of Ground Rules From Table 4-3 Used to Determine Which Propulsion and Power Technologies Would be Evaluated for ORM 1 (GEO)

Move \ Hold	ch	cy	nb	st	ne	se
ch	6	1	3,5	5	3	PV
cy	1,6	1	1,3,5	1,5	1,3	1
nb	3,4,6	1,3,4	3,4,5	2,3,4,5	3,4	2,3,4
st	4,6	1,4	2,3,4,5	4,5	2,3,4	4
ne	3,6	1,3	3,5	2,3,5	3	2,3
se	6	1	2,3,5	5	2,3	PV

Chemical Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	6	1	3,5	5	3	PV
cy	1,6	1	1.3.5	1,5	1.3	1
nb	3,4,6	1,3,4	3,4,5	2,3,4,5	3,4	2,3,4
st	4,6	1,4	2,3,4,5	4,5	2,3,4	4
ne	3,6	1,3	3,5	2,3,5	3	2,3
se	6	1	2,3,5	5	2,3	PV

Cryogenic Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	6	1	5	2,5	NB	2
cy	1,6	1	1,5	1,2,5	1	1,2
nb	6	1	5	2,5	NB	2
st	2,6	1,2	2,5	2,5	2	2
ne	6	1	5	2,5	NB	2
se	2,6	1,2	2,5	2,5	2	2

Nuclear Bimodal Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	6	1	2,3,5	5	2,3	SB/PV
cy	1,6	1	1,2,3,5	1,5	1,2,3	1
nb	2,3,6	1,2,3	2,3,5	2,3,5	2,3	2,3
st	6	1	2,3,5	5	2,3	SB
ne	2,3,6	1,2,3	2,3,5	2,3,5	2,3	2,3
se	6	1	2,3,5	5	2,3	SB/PV

Solar Thermal Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	6	1	5	2,5	NE	2
cy	1,6	1	1,5	1,2,5	1	1,2
nb	4,6	1,4	4,5	2,4,5	4	2,4
st	2,4,6	1,2,4	2,4,5	2,4,5	2,4	2,4
ne	6	1	5	2,5	NE	2
se	2,6	1,2	2,5	2,5	2	2

Nuclear Electric Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	6	1	2,3,5	5	2,3	PV
cy	1,6	1	1,2,3,5	1,5	1,2,3	1
nb	2,3,4,6	1,2,3,4	2,3,4,5	2,3,4,5	2,3,4	2,3,4
st	4,6	1,4	2,3,4,5	4,5	2,3,4	4
ne	2,3,6	1,2,3	2,3,5	2,3,5	2,3	2,3
se	6	1	2,3,5	5	2,3	PV

Solar Electric Lift

ch = baseline chemical
nb = nuclear bimodal
ne = nuclear electric
cy = advanced cryogenic
st = solar thermal/bimodal
se = solar electric
NB = nuclear bimodal
PV = photovoltaic
NE = nuclear electric
SB = solar bimodal

Table 4-5. Matrix of Ground Rules From Table 4-3 Used to Determine Which Propulsion and Power Technologies Would be Evaluated for ORM 2 (GPS)

Chemical Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	PV	1,7,9	3,7,5,9	7,5,9	3,7,9	7,9
cy	1,7,9	1,7	1,3,7,5,9	1,7,5,9	1,3,7,9	1,7,9
nb	3,4,7,9	1,3,4,7,9	3,4,7,5	2,3,4,7,5,9	3,4,7,9	2,3,4,7,9
st	4,7,9	1,4,7,9	2,3,4,7,5,9	4,7,5	2,3,4,7,9	4.7,9
ne	3,7,9	1,3,7,9	3,7,5,9	2,3,7,5,9	3,7	2,3,7,9
se	7,9	1,7,9	2,3,7,5,9	7,5,9	2,3,7,9	7

Cryogenic Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	PV	1,7,9	3,7,5,9	7,5,9	3,7,9	7,9
cy	1,7,9	1,7	1,3,7,5,9	1,7,5,9	1,3,7,9	1,7,9
nb	3,4,7,9	1,3,4,7,9	3,4,7,5	2,3,4,7,5,9	3,4,7,9	2,3,4,7,
st	4,7,9	1,4,7,9	2,3,4,7,5,9	4,7,5	2,3,4,7,9	4,7,9
ne	3,7,9	1,4,7,9	2,3,4,7,5,9	4,7,5,9	2,3,4,7	2,3,7,9
se	7,9	1,7,9	2,3,7,5,9	7,5,9	2,3,7,9	7

Nuclear Bimodal Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	NB	1,7,9	7,5,9	2,7,5,9	7,9	2,7,9
cy	1,7,9	1,7	1,7,5,9	1,2,7,5,9	1,7,9	1,2,7,9
nb	7,9	1,7,9	7,5	2,7,5,9	7,9	2,7,9
st	2,7,9	1,2,7,9	2,7,5,9	2,7,5	2,7,9	2,7,9
ne	7,9	1,7,9	7,5,9	2,7,5,9	7	2,7,9
se	2,7,9	1,2,7,9	2,7,5,9	2,7,5,9	2,7,9	2,7

Solar Thermal Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	SB/PV	1,7,9	2,3,7,5,9	7,5,9	2,3,7,9	7,9
cy	1,7,9	1,7	1,2,3,7,5,9	1,7,5,9	1,2,3,7	1,7,9
nb	2,3,7,9	1,2,3,97	2,3,7,5	2,3,7,5,9	2,3,7,9	2,3,7,9
st	7,9	1,7,9	2,3,7,5,9	7,5	2,3,7,9	7,9
ne	2,3,7,9	1,2,3,7	2,3,7,5,9	2,3,7,5,9	2,3,7	2,3,7,9
se	7,9	1,7,9	2,3,7,5,9	7,5,9	2,3,7,9	7

Nuclear Electric Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	NE	1,7,9	7,5,9	2,7,5,9	7,9	2,7,9
cy	1,7,9	1,7	1,7,5,9	1,2,7,5,9	1,7,9	1,2,7,9
nb	4,7,9	1,4,7,9	4,7,5	2,4,7,5	4,7,9	2,4,7,9
st	2,4,7,9	1,2,4,7,9	2,4,7,5,9	2,4,7,5	2,4,7,9	2,4,7,9
ne	7,9	1,7,9	7,5,9	2,7,5,9	7	2,7,9
se	2,7,9	1,2,7,9	2,7,5,9	2,7,5,9	2,7,9	2,7

Solar Electric Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	PV	1,7,9	2,3,7,5,9	7,5,9	2,3,7,9	7,9
cy	1,7,9	1,7	1,2,3,7,5,9	1,7,5,9	1,2,3,7,9	1,7,9
nb	2,3,4,7,9	1,2,3,4,7,9	2,3,4,7,5	2,3,4,7,5,9	2,3,4,7,9	2,3,4,7,9
st	4,7,9	1,4,7,9	2,3,4,7,5,9	4,7,5	2,3,4,7,9	4,7,9
ne	2,3,7,9	1,2,3,7,9	2,3,7,5,9	2,3,7,5,9	2,3,7	2,3,7,9
se	7,9	1,7,9	2,3,7,5,9	7,5,9	2,3,7,9	7

ch = baseline chemical
nb = nuclear bimodal
ne = nuclear electric
cy = advanced cryogenic
st = solar thermal/bimodal
se = solar electric
NB = nuclear bimodal
PV = photovoltaic
NE = nuclear electric
SB = solar bimodal

Table 4-6. Matrix of Ground Rules From Table 4-3 Used to Determine Which Propulsion and Power Technologies Would be Evaluated for ORM 3a,b (LEO)

Chemical Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	PV	1,7,9	3,7,5,9	7,5,9	3,7,9	7,9
cy	1,7,9	1,7	1,3,7,5,9	1,7,5,9	1,3,7,9	1,7,9
nb	3,4,7,9	1,3,4,7,9	3,4,7,5	2,3,5,4,7,9	3,4,7,9	2,3,4,7,9
st	4,7,9	1,4,7,9	2,3,4,7,5,9	4,7,5	2,3,4,7,9	4,7,9
ne	3,7,9	1,3,7,9	3,7,5,9	2,3,7,5,9	3,7	2,3,7,9
se	7,9	1,7,9	2,3,7,5,9	7,5,9	2,3,7,9	7

Cryogenic Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	PV	1,7,9	3,7,5,9	7,5,9	3,7,9	7,9
cy	1,7,9	1,7	1,3,7,5,9	1,7,5,9	1,3,7,9	1,7,9
nb	3,4,7,9	1,3,4,7,9	3,4,7,5	2,3,4,7,5,9	3,4,7,9	2,3,4,7,9
st	4,7,9	1,4,7,9	2,3,4,7,5,9	4,7,5	2,3,4,7,9	4,7,9
ne	3,7,9	1,3,7	3,7,5,9	2,3,7,5,9	3,7	2,3,7,9
se	7,9	1,7,9	2,3,7,5,9	7,5,9	2,3,7,9	7

Nuclear Bimodal Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	8	1,7,8,9	7,8,5	2,7,8,5,9	7,8,9	2,7,8,9
cy	1,7,8,9	1,7,8	1,7,8,5,9	1,2,7,8,5,9	1,7,8,9	1,2,7,8,9
nb	7,8,9	1,7,8,9	7,8,5	1,7,8,5,9	7,8,9	2,7,8,9
st	2,7,8,9	1,2,7,8,9	2,7,8,5,9	2,7,8,5	2,7,8,9	2,7,8,9
ne	7,8,9	1,7,8,9	7,8,5,9	2,7,8,5,9	7,8	2,7,8,9
se	2,7,8,9	1,2,7,8,9	2,7,8,5,9	2,7,8,5,9	2,7,8,9	2,7,8

Solar Thermal Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	8	1,7,8,9	2,3,7,8,5,9	7,8,5,9	2,3,7,8,9	7,8,9
cy	1,7,8,9	1,7,8	1,2,3,7,8,5,9	1,7,8,5,9	1,2,3,7,8,9	1,7,8,9
nb	2,3,7,8,9	1,2,3,7,8,9	2,3,7,8,5	2,3,7,8,5,9	2,3,7,8,9	2,3,7,8,9
st	7,8,9	1,7,8,9	2,3,7,8,5,9	7,8,5	2,3,7,8,9	7,8,9
ne	2,3,7,8,9	1,2,3,7,8,9	2,3,7,8,5,9	2,3,7,8,5,9	2,3,7,8	2,3,7,8,9
se	7,8,9	1,7,8,9	2,3,7,8,5,9	7,8,5,9	2,3,7,8,9	7,8

Nuclear Electric Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	8	1,7,8,9	7,8,5,9	2,7,8,5,9	7,8,9	2,7,8,9
cy	1,7,8,9	1,7,8	1,7,8,5,9	1,2,7,8,5,9	1,7,8,9	1,2,7,8,9
nb	4,7,8,9	1,4,8,7,9	4,7,8,5	2,4,7,8,5,9	4,7,8,9	2,7,8,9
st	2,4,7,8,9	1,2,4,7,8,9	2,4,7,8,5,9	2,4,7,8,5	2,4,7,8,9	2,4,7,8,9
ne	7,8,9	1,7,8,9	7,8,5,9	2,7,8,5,9	7,8	2,7,8,9
se	2,7,8,9	1,2,7,8,9	2,7,8,5,9	2,7,8,5,9	2,7,8,9	2,7,8

Solar Electric Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	PV	1,7,9	2,3,7,5,9	7,5,9	2,3,7,9	7,9
cy	1,7,9	1,7	1,2,3,7,5,9	1,7,5,9	1,2,3,7,9	1,7,9
nb	2,3,4,7,9	1,2,3,4,7,9	2,3,4,7,5	2,3,4,7,5,9	2,3,4,7,9	2,3,4,7,9
st	4,7,9	1,4,7,9	2,3,4,7,5,9	4,7,5	2,3,4,7,9	4,7,9
ne	2,3,7,9	1,2,3,7,9	2,3,7,5,9	2,3,7,5,9	2,3,7	2,3,7,9
se	7,9	1,7,9	2,3,7,5,9	7,5,9	2,3,7,9	7

ch = baseline chemical
cy = advanced cryogenic
nb = nuclear bimodal
st = solar thermal/bimodal
ne = nuclear electric
se = solar electric
NB = nuclear bimodal
NE = nuclear electric
PV = photovoltaic
SB = solar bimodal

Table 4-7. Matrix of Ground Rules From Table 4-3 Used to Determine Which Propulsion and Power Technologies Would be Evaluated for ORM 4 (HEO)

Chemical Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	PV	1,13	3,8,13	8,13	3,13	13
cy	1,13	1	1,3,8,13	1,8,13	1,3,13	1,13
nb	3,4,13	1,3,4,13	3,4,8	2,3,4,8,13	3,4,13	2,3,4,13
st	4,13	1,4,13	2,3,4,8,13	4,8	2,3,4,13	4,13
ne	3,13	1,3,13	3,8,13	2,3,8,13	3	2,3,13
se	13	1,13	2,3,8,13	8,13	2,3,13	PV

Cryogenic Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	PV	1,13	3,8,13	8,13	3,13	13
cy	1,13	1	1,3,8,13	1,8,13	1,3,13	1,13
nb	3,4,13	1,3,4,13	3,4,8	2,3,4,8,13	3,4,13	2,3,4,13
st	4,13	1,4,13	2,3,4,8,13	4,8	2,3,4,13	4,13
ne	3,13	1,3,13	3,8,13	2,3,8,13	3	2,3,13
se	13	1,13	2,3,8,13	8,13	2,3,13	PV

Nuclear Bimodal Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	NB	1,13	8,13	2,8,13	13	2,13
cy	1,13	1	1,8,13	1,2,8,13	1,13	1,2,13
nb	13	1,13	8	2,8,13	13	2,13
st	2,13	1,2,13	2,8,13	2,8	2,13	2,13
ne	13	1,13	8,13	2,8,13	NB	2,13
se	2,13	1,2,13	2,8,13	2,8,13	2,13	2

Solar Thermal Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	SB/PV	1,13	2,3,8,13	8,13	2,3,13	13
cy	1,13	1	1,2,3,8,13	1,8,13	1,2,3,13	1,13
nb	2,3,13	1,2,3,13	2,3,8	2,3,8,13	2,3,13	2,3,13
st	13	1,13	2,3,8,13	8	2,3,13	13
ne	2,3,13	1,2,3,13	2,3,8,13	2,3,8,13	2,3	2,3,13
se	13	1,13	2,3,8,13	8,13	2,3,13	SB/PV

Nuclear Electric Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	12,13	1,12,13	8,12,13	2,8,12,13	12,13	2,12,13
cy	1,12,13	1,12	1,8,12,13	1,2,8,12,13	1,12,13	1,2,12,13
nb	4,12,13	1,4,12,13	4,8,12	2,4,8,12,13	4,12,13	2,4,12,13
st	2,4,12,13	1,2,4,12,13	2,4,8,12,13	2,4,8,12	2,4,12,13	2,4,12,13
ne	12,13	1,12,13	8,12,13	2,8,12,13	12	2,12,13
se	2,12,13	1,2,12,13	2,8,12,13	2,8,12,13	2,12,13	2,12,

Solar Electric Lift

Move \ Hold	ch	cy	nb	st	ne	se
ch	12	1,12,13	2,3,8,12,13	8,12,13	2,3,12,13	12,13
cy	1,12,13	1,12	1,2,3,8,12,13	1,8,12,13	1,2,3,12,13	1,12,13
nb	2,3,4,12,13	1,2,3,4,12,1	2,3,4,8,12	2,3,4,8,12,13	2,3,4,12,13	2,3,4,12,13
st	4,12,13	1,4,12,13	2,3,4,8,12,13	4,8,12	2,3,4,12,13	4,12,13
ne	2,3,12,13	1,2,3,12,13	2,3,8,12,13	2,3,8,12,13	2,3,12	2,3,12,13
se	12,13	1,12,13	2,3,8,12,13	8,12,13	2,3,12,13	12

ch = baseline chemical
nb = nuclear bimodal
ne = nuclear electric
cy = advanced cryogenic
st = solar thermal/bimodal
se = solar electric
NB = nuclear bimodal
PV = photovoltaic
NE = nuclear electric
SB = solar bimodal

Table 4-8. Combinations of Lift, Hold, Move, and Electric Power Technologies Evaluated in the OECS

No.	Category	Lift	Hold	Move	Electric Power
ORM 1. GEO					
1-1	Baseline	Baseline Chem[a]	N_2H_4 Arcjet	N_2H_4 Arcjet	Photovoltaic
1-2		Baseline Chem[a]	N_2H_4 Arcjet	Biprop Chem	Photovoltaic
1-3	Advanced Cryo	Advanced Cryo[a]	N_2H_4 Arcjet	N_2H_4Arcjet	Photovoltaic
1-4		Advanced Cryo[a]	N_2H_4 Arcjet	Biprop Chem	Photovoltaic
1-5	Nuclear Bimodal	H_2 Nucl Bimodal	N_2H_4 Arcjet	N_2H_4 Arcjet	Thermoelectric
1-6		H_2 Nucl Bimodal	N_2H_4 Arcjet	Monoprop Chem	Thermoelectric
1-7		H_2 Nucl Bimodal	N_2H_4 Arcjet	NH_3 Nucl Bimodal	Thermoelectric
1-8	Solar Bimodal	H_2 Solar Bimodal	N_2H_4 Arcjet	N_2H_4 Arcjet	Thermionic
1-9		H_2 Solar Bimodal	N_2H_4 Arcjet	Monoprop Chem	Thermionic
1-10		H_2 Solar Bimodal	N_2H_4 Arcjet	NH_3 Solar Bimodal	Thermionic
1-11	Solar Thermal	H_2 Solar Thermal	N_2H_4 Arcjet	N_2H_4 Arcjet	Photovoltaic
1-12		H_2 Solar Thermal	N_2H_4 Arcjet	Monoprop Chem	Photovoltaic
1-13	Nuclear Electric	NH_3 Arcjet	NH_3 Arcjet	NH_3 Arcjet	Thermionic
1-14		NH_3 Arcjet	NH_3 Arcjet	Monoprop Chem	Thermionic
1-15		H_2 Arcjet	N_2H_4 Arcjet	N_2H_4 Arcjet	Thermionic
1-16		H_2 Arcjet	N_2H_4 Arcjet	Monoprop Chem	Thermionic
1-17		Xe SPT	Xe SPT	Xe SPT	Thermionic
1-18		Xe SPT	Xe SPT	Monoprop Chem	Thermionic
1-19		Xe Ion	Xe Ion	Xe Ion	Thermionic
1-20		Xe Ion	Xe Ion	Monoprop Chem	Thermionic
1-21	Solar Electric	NH_3 Arcjet	NH_3 Arcjet	NH_3 Arcjet	Photovoltaic
1-22		NH_3 Arcjet	NH_3 Arcjet	Monoprop Chem	Photovoltaic
1-23		H_2 Arcjet	N_2H_4 Arcjet	N_2H_4 Arcjet	Photovoltaic
1-24		H_2 Arcjet	N_2H_4 Arcjet	Monoprop Chem	Photovoltaic
1-25		Xe SPT	Xe SPT	Xe SPT	Photovoltaic
1-26		Xe SPT	Xe SPT	Monoprop Chem	Photovoltaic
1-27		Xe Ion	Xe Ion	Xe Ion	Photovoltaic
1-28		Xe Ion	Xe Ion	Monoprop Chem	Photovoltaic

[a]For Titan IV, listed technology places the payload into GEO. For Atlas and Delta, listed technology only goes as far as Geosynchronous Transfer Orbit (GTO); an integral propulsion subsystem using bipropellant (N_2H_4 and N_2O_4) provides final orbital circularization.

Table 4-8. (continued)

No.	Category	Lift	Hold	Move	Electric Power
ORM 2a. MEO-GPS and ORM 2b. MEO-Low					
2-1	Baseline	Baseline Chem[b]	Monoprop Chem	N/A	Photovoltaic
2-2	Advanced Cryo	Advanced Cryo[b]	Monoprop Chem	N/A	Photovoltaic
2-3	Nuclear Bimodal	H_2 Nucl Bimodal	Monoprop Chem	N/A	Thermoelectric
2-4	Solar Bimodal	H_2 Solar Bimodal	Monoprop Chem	N/A	Thermionic
2-5	Solar Thermal	H_2 Solar Thermal	Monoprop Chem	N/A	Photovoltaic
2-6	Nuclear Electric	NH_3 Arcjet	Monoprop Chem	N/A	Thermionic
2-7		H_2 Arcjet	Monoprop Chem		Thermionic
2-8		Xe SPT	Monoprop Chem		Thermionic
2-9		Xe Ion	Monoprop Chem		Thermionic
2-10	Solar Electric	NH_3 Arcjet	Monoprop Chem	N/A	Photovoltaic
2-11		H_2 Arcjet	Monoprop Chem		Photovoltaic
2-12		Xe SPT	Monoprop Chem		Photovoltaic
2-13		Xe Ion	Monoprop Chem		Photovoltaic
ORM 3a. LEO-Polar and ORM 3b. LEO-Big					
3-1	Baseline	Direct Insertion[c]	Monoprop Chem	N/A	Photovoltaic
3-2		Biprop Chem	Monoprop Chem		Photovoltaic
3-3	Advanced Cryo	Advanced Cryo	Monoprop Chem	N/A	Photovoltaic
3-4		Adv Cryo plus Biprop[d]	Monoprop Chem		Photovoltaic
3-5	Solar Electric	H_2 Arcjet	Monoprop Chem	N/A	Photovoltaic
3-6		NH_3 Arcjet	Monoprop Chem		Photovoltaic
3-7		N_2H_4 Arcjet	Monoprop Chem		Photovoltaic
3-8		Xe SPT	Monoprop Chem		Photovoltaic
3-9		Xe Ion	Monoprop Chem		Photovoltaic

[b]For Titan IV, listed technology places the payload into MEO. For Atlas and Delta, listed technology only goes as far as an elliptical transfer orbit; an integral propulsion subsystem using bipropellant (N_2H_4 and N_2O_4) provides final orbital circularization.

[c]Direct insertion by the launch vehicle; no upper stage is used.

[d]Advanced cryo upper stage for transfer orbit; bipropellant integral propulsion subsystem to circularize.

Table 4-8. (concluded)

No.	Category	Lift	Hold	Move	Electric Power
			ORM 4. HEO		
4-1 4-2	Baseline	Direct Insertion[e] Biprop Chem[e]	N_2H_4 Arcjet Monoprop Chem	N/A	Photovoltaic
4-3 4-4	Advanced Cryo	Advanced Cryo[f]	Monoprop Chem N_2H_4 Arcjet	N/A	Photovoltaic
4-5 4-6	Nuclear Bimodal	H_2 Nucl Bimodal	Monoprop Chem N_2H_4 Arcjet	N/A	Thermoelectric
4-7 4-8	Solar Bimodal	H_2 Solar Bimodal	Monoprop Chem N_2H_4 Arcjet	N/A	Thermionic
4-9 4-10	Solar Thermal	H_2 Solar Thermal	Monoprop Chem N_2H_4 Arcjet	N/A	Photovoltaic

[e]Atlas and Delta insert the satellite directly into HEO. Titan no upper stage inserts the satellite into an orbit whose perigee is low and is subsequently raised by an integral bipropellant subsystem (N_2H_2 and N_2O_2).
[f]An advanced cryogenic upper stage is not applicable to Titan (see previous note).

Table 4-9. Propulsion Subtechnologies Considered in the OECS

	Lift Sub-technologies	Hold/Move Sub-technologies[a,b]	Remarks
Baseline Chemical	Lift subtechnology inherent in current upper stages	Monoprop Biprop N_2H_4 Arcjet	Photovoltaic electric power only. All subtechnologies state of the art. Monoprop for low ΔV hold/move, biprop or N_2H_4 arcjet otherwise.
Cryogenic	LO_2/ LH_2 Biprop	Monoprop Biprop N_2H_4 Arcjet	Photovoltaic electric power only. All hold/move subtechnologies state of the art. Monoprop for low ΔV hold/move, biprop or N_2H_4 arcjet otherwise.
Nuclear Bimodal	H_2 Nuclear Bimodal (Thermal)	NH_3 Thermal (move only) N_2H_4 Arcjet Monoprop	Nuclear bimodal electric power only. Thermal propulsion for moderate or high ΔV move only. Monoprop for low ΔV hold/move and fast moves.

Table 4-9. (concluded)

	Lift Sub-technologies	Hold/Move Sub-technologies[a,b]	Remarks
Solar Bimodal	H_2 Thermal	NH_3 Thermal (move only) N_2H_4 Arcjet Monoprop	Solar bimodal electric power only. Thermal propulsion for moderate or high ΔV move only. Monoprop for low ΔV hold/move and fast moves.
Solar Thermal	H_2 Thermal	N_2H_4 Arcjet NH_3 Arcjet Monoprop	Solar thermal system discarded after lift. Photovoltaic electric power only. Monoprop for low ΔV hold/move and fast moves.
Nuclear Electric	N_2H_4 Arcjet NH_3 Arcjet H_2 Arcjet SPT Ion	N_2H_4 Arcjet NH_3 Arcjet SPT Ion Monoprop	Nuclear electric power only. Monoprop for low ΔV hold/move and quick moves.
Solar Electric	N_2H_4 Arcjet NH_3 Arcjet H_2 Arcjet SPT Ion	N_2H_4 Arcjet NH_3 Arcjet SPT Ion Monoprop	Photovoltaic electric power only. Monoprop for low ΔV hold/move and quick moves.

[a] H_2 arcjet requires LH_2 which is not suitable for hold/move.
[b] SPT and ion hold/move only when used for lift.

Table 4-10. Subtechnologies Selected as a Function of Hold and Move Requirements

	Low Hold ΔV (ORMs 2, 3a,b, and 4)	Moderate Hold ΔV (ORM 1)	Low Maneuver ΔV (ORMs 2, 3a,b, and 4)	Moderate Maneuver ΔV (ORM 1)
State of the art technologies	Cold Gas or Monoprop	Biprop and N_2H_4 Arcjet	N/A (essentially no maneuver requirements)	N_2H_4 (low thrust, high I_{SP}) or monoprop or bi-prop (high thrust, low I_{SP})
Additional technologies considered	None	SPT and Ion	N/A (essentially no maneuver requirements)	SPT, ion, and NH_3 bimodal lift technologies

4.3.9 Astrodynamics

4.3.9.1 INTRODUCTION

Propulsive thrust levels vary orders of magnitude among the OECS lift technologies: from more than one hundred thousand newtons for the advanced cryogenic upper stages to a few newtons for electric thrusters. To guarantee fair technology comparisons over this range, we must find advantageous launch and transfer scenarios for each technology and ORM. In practical terms, this requires carefully selecting values for those parameters that affect launch and subsequent transfer to the final mission orbit.

Baseline and advanced cryogenic systems have fixed thrust and are assumed to launch into ORM-specific orbits independent of the mass to be delivered. For example, the Delta II and Atlas IIAS vehicles launch directly into a geosynchronous transfer orbit (GTO) for the GEO ORM. For all other technologies, drop-off (separation) of the satellite with its lift propulsion from the booster takes place in a parking orbit. In the OECS, the lowest possible parking orbit for a given technology is assumed to be circular, with its altitude determined from considerations of atmospheric drag. Higher parking orbits are assumed to be elliptical, with perigee at the altitude of the circular parking orbit. For the electric propulsion technologies, drop-off is assumed always to be into a circular parking orbit. The thermal propulsion systems (nuclear and solar bimodal and solar thermal) were evaluated for circular and elliptical parking orbits, with the parking orbit selected to maximize the payload on orbit or minimize the propulsion cost of transfer. Parking orbits are illustrated in Figure 4-3. A parking orbit is characterized by its inclination and apogee altitude, which are determined by the launch vehicle. Transfer conditions for the technologies considered in OECS are discussed in detail later in this section.

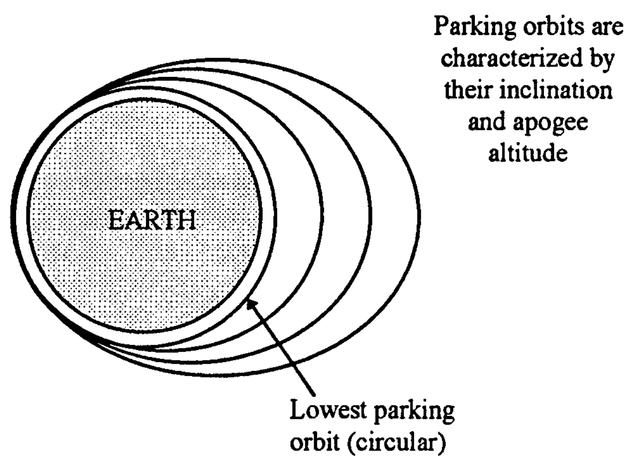


Figure 4-3. Possible drop-off orbits.

We have three lift propulsion situations to consider:

- Completely fixed designs: baseline and advanced cryogenic systems
- Fixed thrust: nuclear bimodal system
- Scalable designs with variable thrust and mass: electric, solar bimodal, and solar thermal systems

For the fixed designs, the transfer strategy includes direct injection by the launch vehicle and, if necessary, an apogee maneuver performed by an apogee kick motor. For the fixed thrust case, we must specify drop-off conditions and how the propulsion is to be applied. This latter is the "burn strategy," and it dictates the locations and lengths of thrusting. For the scalable designs, not only drop-off conditions and burn strategy must be considered: thrust level and trip time—the time needed to reach mission orbit from drop-off—must be considered as well. The scalable designs present the most complex considerations.

Later in this section we make generic decisions about drop-off conditions and burn strategies. Having done this, we are in position to select trip time and determine the necessary thrust level *or* select thrust level and determine the trip time. We have uniformly done the former: select trip time.

4.3.9.2 LAUNCH

Launches from both Vandenburg AFB and Cape Canaveral AS are constrained in azimuth to avoid overflight of land. Within these constraints, however, a launch vehicle can place a satellite and transfer vehicle into a range of inclinations. The best inclination is the one that allows the transfer vehicle to place the maximum satellite payload mass in final orbit. Finding this inclination involves a trade-off of mass in parking orbit and the ΔV that must be applied to that mass by the lift propulsion to perform the transfer.

Two scenarios must be compared: (1) launch on the minimum energy trajectory (e.g., due east from Cape Canaveral) and perform the necessary plane change with the transfer vehicle; (2) launch to minimize the plane change accomplished with the transfer vehicle. The first case allows the maximum mass to be placed into a given drop-off orbit. Because there is no opportunity to make design trade-offs for a fixed-design transfer vehicle, the straightforward answer is to minimize the plane change required of the transfer vehicle within launch vehicle constraints. However scalable transfer vehicles allow a trade-off between the mass of the lift system placed on orbit and the amount of plane change it accomplishes.

There is never a trade-off for transfer to ORM 1 at GEO because launching due east from Cape Canaveral also minimizes the transfer vehicle's plane change. Trade-offs are possible with other ORMs. In these other cases, we have asserted that the maximum payload mass in mission orbit is obtained from the second alternative: minimizing the plane change performed by the transfer vehicle. To prove this assertion would require examining each valid combination of ORM, launch complex, launch vehicle, and transfer technology. Instead, we examined a case whose parameters are known to be unfavorable to our assertion, and we generalized from that result. The case we selected is a transfer from LEO to ORM 2a (GPS) launched from Cape Canaveral on an Atlas IIAS with an electric ion propulsion lift system. This case supported our assertion.

4.3.9.3 LIFT (ORBITAL TRANSFER)

To structure our lift discussion, we group the technologies according to thrust magnitude as follows:

- High thrust: baseline chemical and advanced cryogenic (impulsive)
- Moderate thrust: nuclear bimodal, solar bimodal, and solar thermal (multi-thrust)
- Low thrust: nuclear electric and solar electric (continuous thrust)

For the bottom-up effectiveness analysis, the transfer vehicle is sized to maximize payload in mission orbit. For the top-down cost-effectiveness analysis, the transfer vehicle is sized to minimize cost for specific mission payloads subject to the constraints, which will be discussed. Minimum cost is always achieved with the smallest launch vehicle.

For both the baseline chemical and advanced cryogenic upper stages, all burns are essentially impulsive and transfer ΔVs are essentially theoretical minimums. Drop-off altitude, thrust level, and trip time are dictated by the launch vehicle and the upper stages. These values are the same for the bottom-up or top-down analyses.

Minimum drop-off altitude. For the moderate-thrust cases, a minimum drop-off altitude of 185 km (100 nm) is employed. Atmospheric drag is high at this altitude, but the moderate-thrust system will quickly raise the orbit and leave the high-drag regime. For the low-thrust cases, a minimum drop-off altitude of 370 km (200 nm) is employed. The altitude of 370 km is supported by an analysis prepared by McLain and Sutton (Zondervan, pp.1-15 to 1–19).

Table 4-13 and Table 4-14 present some results for a planned excursion looking at a drop-off altitude of 700 km for the solar electric and nuclear technologies. However, the relevant effectiveness and cost-effectiveness analyses were not performed.

Burn Strategies. For moderate-thrust cases, the burns are most efficiently applied within a few degrees of either perigee or apogee. These burns raise the orbit, circularize it, and make necessary plane change. Using multiple perigee and apogee burns is the most propellant-efficient approach, because it reduces "gravity losses" when thrusting is not perpendicular to the orbital radius vector. (Gravity losses are discussed on pages 146-152 of Chobotov.) Based on our investigation, the most efficient burn strategy for ORMs 1 and 2 is to burn repeatedly at two points: first, at perigee to simultaneously raise apogee and perform a small amount of plane change and, second, at apogee to simultaneously raise perigee and perform the bulk of any necessary plane change. Moderate-thrust ΔVs are only slightly larger than impulsive ΔVs when this strategy is employed.

For low-thrust cases, the burn strategy is to thrust continuously, spiraling out to raise the orbit and simultaneously performing necessary plane changes. This strategy minimizes trip times but is subject to large gravity losses. ΔVs for low-thrust cases are given accurately with the Edelbaum approximation (Chobotov, pp. 152–153). They are significantly higher than corresponding impulsive ΔVs, especially with respect to the plane change. Low thrust is not suited to highly eccentric ORM 4 because of high gravity losses and very long trip times. Low thrust was not evaluated for ORM 4.

Trip Times. We have explained that it is necessary to fix either trip time or thrust to determine a design. Figure 4-4 illustrates the qualitative nature of the trade-offs. For moderate or low-thrust technologies, higher thrust means shorter trip times but also higher lift propulsion cost because of increased electrical or thermal energy requirements.

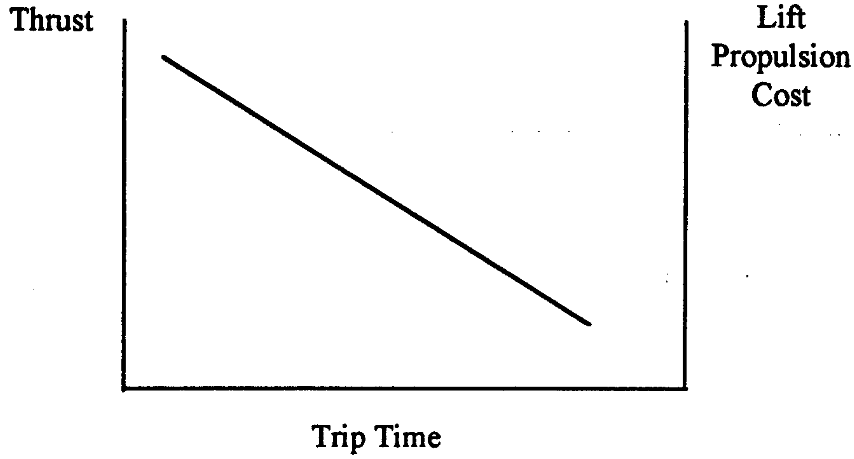


Figure 4-4. Trend of the relationships among trip time, thrust, and cost.

There is a relationship between trip time and payload mass in mission orbit as well. The nature of this relationship is illustrated in Figure 4-5. Payload mass increases with trip time. Combining the relationships of Figure 4-4 and Figure 4-5 tells us that longer trip times increase payload in mission orbit and result in a lower-thrust, less costly transfer vehicle. There is, however, another important cost consideration: longer trip times increase the need for on-orbit spare satellites to maintain high constellation availability (as discussed in Section 4.3.5). Maintaining spares adds to constellation maintenance costs.

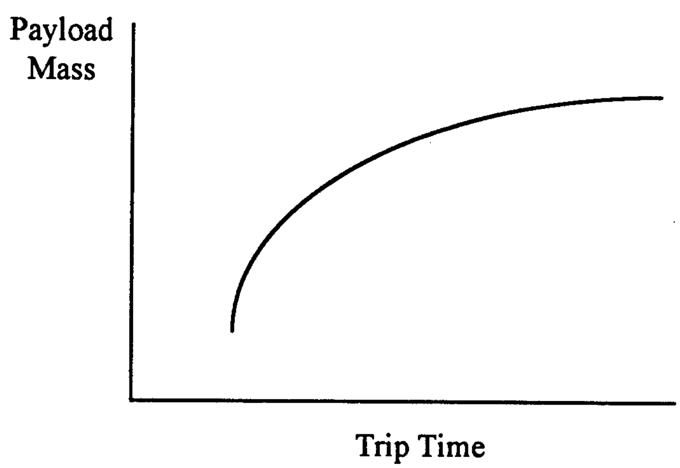


Figure 4-5. Trend of the relationships between trip time and payload mass.

How do we strike a balance between trip time and constellation maintenance cost? We find an answer in the *Solar Electric Propulsion Assessment* (Chan et al.) and extensions of that work in the OECS. For ORM 1 (GEO), the *Solar Electric Propulsion Assessment* provides constellation availabilities for 3- and 5-satellite constellations as a function of the number of on-orbit spares and the deployment time (the time from call-up to having a functioning satellite). These results show the number of satellites needed to support a typical 3- or 5-satellite constellation at a minimum 0.9 availability is essentially independent of deployment time for deployment times of 0–90 days. By using the thrust level of the thermal propulsion systems and allowing 60 days for launch and checkout, we get a nominal trip time to GEO of 30 days. We examine a 60-day excursion as well. Shorter trip times are specified for other ORMs based on their lower ΔV requirements.

Typical trip times to GEO using electric propulsion are six months or longer for reasonable power levels (Chan et al., Table 5-17). Unlike the case for the relatively short trip times of thermal systems, the long trip times for electric propulsion systems can significantly affect the number of satellites needed to maintain a constellation over its life (and, thus, the cost).

For very long trip times, it is impossible to maintain a constellation at high availability with electric propulsion. For example, Table 5-9 from the *Solar Electric Propulsion Assessment* (Chan et al., p. 5-37) shows that a 0.9 constellation availability for a 5-satellite GEO constellation with a spare cannot be maintained even in the case of an extended life satellite (MMD = 10.0 yr) for deployment as short as a year. Similarly, two spares and a deployment time of 480 days fall short of 0.9 availability. On the other hand, we note from the same table that for shorter deployment times, a spare is still necessary for as few as 90 days deployment time. Since the number of satellites required by the table is nearly constant over the 90–360-day deployment range, cost is nearly constant as well except for the cost of increasing electrical power to achieve the shorter trip times. (It is true that shorter trip times allow a higher constellation availability, but we are comparing equal availabilities.) From this, we can conclude that deployment times in excess of one year are not desirable from a cost-effectiveness—and, probably, an operational—point of view.

The trip times used in the study are shown in Table 4-11. Assuming a maximum deployment time of 360 days and 60 days for launch and on-orbit checkout, we have selected a trip time for electric propulsion systems of 300 days for ORM 1 (GEO), the ORM requiring the highest ΔV. Trip times for ORM 2 (GPS) are lower based on their lower ΔV requirements. Trip times for LEO ORMs are short and have been chosen to keep propulsion power levels low.

Table 4-11. Summary by Technology of Orbital Transfer Parameters

Technology	Drop-off Altitude	Thrust Level	Trip Time ORM 1	Trip Time ORM 2	Trip Time ORM 3a,b	Trip Time ORM 4
Bottom-Up (Effectiveness Analysis)						
Baseline	Varies	Fixed	~1 day	~1 day	~1 day	~1 day
Advanced Cryo	Varies	Fixed	~1 day	~1 day	~1 day	~1 day
Nuclear Bimodal	185 km	Fixed	<40 hr[a]	<40 hr[a]	N/A	<40 hr
Solar Bimodal	185 km	Calculated	30 days	25 days	N/A	20 days
Solar Thermal	185 km	Calculated	30 days	25 days	N/A	20 days
Solar Electric	370 km	Calculated	300 days	210 days	60 days	N/A
Nuclear Electric	370 km	Calculated	300 days	210 days	N/A	N/A
Top-Down (Cost-Effectiveness Analysis)						
Baseline	Varies	Fixed	~1 day	~1 day	~1 day	~1 day
Advanced Cryo	Varies	Fixed	~1 day	~1 day	~1 day	~1 day
Nuclear Bimodal	Maximize	Fixed	<40 hr[a]	<40 hr[a]	N/A	<40 hr
Solar Bimodal	Maximize	Minimize	30 days	25 days	N/A	20 days
Solar Thermal	Maximize	Minimize	30 days	25 days	N/A	20 days
Solar Electric	Maximize	Minimize	300 days	210 days	60 days	N/A
Nuclear Electric	Maximize	Minimize	300 days	210 days	N/A	N/A

[a]Fixed thrust results in a variable trip time.

Thrust. Once drop-off conditions, burn strategy, and trip time are fixed, thrust can be determined. The determination is not straightforward and requires the use of sophisticated computer codes. Two codes were used by Rocketdyne: Multiburn for the solar thermal cases, and an OECS system model integrated with the Program to Optimize Simulated Trajectories (POST) (Brauer et al.) for solar and nuclear bimodal cases. The Multiburn code originally written by Phillips Lab is based on *Highly Efficient, Very Low Thrust Transfer to Geosynchronous Orbit: Exact and Approximate Solutions* (Redding). The Multiburn code can determine approximate optimal solutions to orbit transfer problems requiring several successive perigee and apogee burns. The Multiburn code has been modified by Rocketdyne to operate from elliptical starting orbits and to include weight algorithms for absorber/thruster, collector, and propellant tank subsystems. POST is a generalized point-mass, discrete-parameter targeting and optimization program developed for NASA by Martin Marietta. It provides the capability to optimize point mass trajectories for powered or unpowered vehicles operating near Earth. POST's flexible simulation capability is augmented by a discrete parameter optimization capability that includes equality and inequality constraints. A detailed system model that updates the performance of the solar bimodal engine on each orbit of the mission has been integrated with POST. Thrust results from the codes for the Delta II appear in Table 4-12, those for the Atlas IIAS appear in Table 4-13, and those for the Titan IV appear in Table 4-14.

Satellite Deployment Time

Current deployment times are much in excess of 60 days. Ground processing times are as short as 60 days for GPS and as long as approximately one year for DSCS (Adams et al., p. 4-1). To this must be added delivery times (launch to operational satellite) of from 10–25 days for GPS IIR to 80–130 days for DSCS to 200+ days for Milstar (Adams et al., p. 4-9).

However, there is a strong desire to reduce deployment times in the future, and many possibilities for improvement could be implemented. An example of potential improvements to DSP processing indicates a reduction from the nominal 170 days to 25 days (Adams et al., p. 4-8).

As always, the sticking point is the trade-off of operational needs and the cost to implement the improvements. We will return to the deployment time question in the cost-effectiveness chapter, Chapter 7.

The tables contain information primarily for the nuclear bimodal, solar bimodal, and solar thermal systems. The data include: trip times; drop-off orbit perigee and apogee; lift thrust; propulsive power, where applicable; drop-off mass; initial thrust-to-weight ratio; total lift ΔV; payload in final mission orbit; and average propulsion system I_{sp}. The nuclear bimodal system has no capability on the Delta II for the assumed power system size and does not appear in Table 4-12.

The nuclear bimodal transfer vehicle has fixed thrust, but the size of its fuel tank can vary with drop-off altitude, resulting in a small variation in cost. Thus its performance is

Table 4-12. Technology Performance Data For Delta II Launch and Transfer to ORM 1 (GEO)

Technology	Trip Time (days)	Initial Perigee (km)	Initial Apogee (km)	Thrust (N)	Propulsive Power (kW)	Initial Mass (kg)	Initial Force/ Weight	ΔV (m/s)	Payload (kg)	Average I_{sp} (s)
Solar Bimodal	30	185	185	283	5.28	5035	0.00574	4323	344	751
	30	185	500	265	4.94	4853	0.00557	4235	356	751
	30	185	700	256	4.77	4740	0.00551	4184	358	751
	30	185	1000	241	4.50	4581	0.00537	4105	363	749
	30	185	1850	214	3.99	4128	0.00529	3910	333	748
	30	185	3700	172	3.21	3520	0.00500	3553	314	741
	30	185	5550	142	2.66	3064	0.00474	3263	308	731
	30	185	7400	124	2.31	2728	0.00464	3049	287	729
	30	185	9250	109	2.04	2472	0.00451	2875	270	728
	30	185	18,500	74	1.37	1792	0.00419	2318	196	721
	60	185	185	140	2.62	5035	0.00284	4318	692	750
	60	185	500	131	2.45	4853	0.00276	4230	694	750
	60	185	700	128	2.38	4740	0.00275	4168	687	748
	60	185	1000	120	2.25	4581	0.00268	4094	682	746
	60	185	1850	107	1.99	4128	0.00264	3894	639	745
	60	185	3700	86	1.61	3520	0.00250	3544	590	739
	60	185	5550	71	1.33	3064	0.00237	3262	536	731
	60	185	7400	62	1.16	2728	0.00232	3047	486	729
	60	185	9250	54	1.02	2472	0.00225	2872	446	727
	60	185	18,500	37	0.69	1792	0.00210	2312	315	719

Table 4-12. (concluded)

Technology	Trip Time (days)	Initial Perigee (km)	Initial Apogee (km)	Thrust (N)	Propul-sive Power (kW)	Initial Mass (kg)	Initial Force/ Weight	ΔV (m/s)	Payload (kg)	Average I_{sp} (s)
Solar Thermal	30	185	185	26.7		5035	0.00054	4729	601	859
	30	185	500	25.7		4853	0.00054	4496	620	858
	30	185	700	25.1		4740	0.00054	4382	621	858
	30	185	1000	24.3		4581	0.00054	4239	616	857
	30	185	1850	21.9		4128	0.00054	3945	561	856
	30	185	3700	18.7		3520	0.00054	3534	469	854
	30	185	5550	16.2		3064	0.00054	3251	376	852
	30	185	7400	14.5		2728	0.00054	3035	300	850
	30	185	9250	13.1		2472	0.00054	2864	237	848
	30	185	18500	9.5		1792	0.00054	2284	59	843
	60	185	185	17.8		5035	0.00036	4535	740	853
	60	185	500	17.2		4853	0.00036	4364	738	853
	60	185	700	16.8		4740	0.00036	4275	730	852
	60	185	1000	16.2		4581	0.00036	4159	715	852
	60	185	1850	14.6		4128	0.00036	3904	641	850
	60	185	3700	12.4		3520	0.00036	3519	533	847
	60	185	5550	10.8		3064	0.00036	3244	431	845
	60	185	7400	9.6		2728	0.00036	3031	349	843
	60	185	9250	8.7		2472	0.00036	2861	281	842
	60	185	18500	6.3		1792	0.00036	2284	92	835

Table 4-13. Technology Performance Data for Atlas IIAS Launch and Transfer to ORM 1 (GEO)

Technology	Trip Time (days)	Initial Perigee (km)	Initial Apogee (km)	Thrust (N)	Propulsive Power (kW)	Initial Mass (kg)	Initial Force/ Weight	ΔV (m/s)	Payload (kg)	Average I_{sp} (s)
Nuclear Bimodal	25.25[a]	185	185	2200		8618	0.0260	4375	1105	820
	22.01[a]	185	500	2200		8392	0.0268	4294	1078	820
	18.90[a]	185	700	2200		8244	0.0272	4261	1050	820
	15.64[a]	185	1000	2200		8051	0.0279	4219	1011	820
	12.73[a]	185	1850	2200		7552	0.0297	4042	937	820
	6.02[a]	185	9250	2200		5330	0.0421	2956	594	820
	26.56[a]	700	700	2200		7881	0.0285	4160	980	820
Solar Bimodal	30	185	185	485	9.04	8618	0.00574	4323	972	751
	30	185	500	458	8.54	8392	0.00557	4235	1009	751
	30	185	700	445	8.29	8244	0.00551	4184	1022	751
	30	185	1000	424	7.90	8051	0.00537	4105	1046	749
	30	185	1850	391	7.30	7552	0.00529	3910	1056	748
	30	185	3700	329	6.14	6722	0.00500	3553	1076	741
	30	185	5550	285	5.31	6124	0.00474	3263	1075	731
	30	185	7400	258	4.81	5674	0.00464	3049	1058	729
	30	185	9250	236	4.39	5330	0.00451	2875	1045	728
	30	185	18,500	178	3.32	4332	0.00419	2318	971	721
	60	185	185	240	4.48	8618	0.00284	4318	1498	750
	60	185	500	227	4.23	8392	0.00276	4230	1505	750
	60	185	700	222	4.15	8244	0.00275	4168	1498	748
	60	185	1000	212	3.95	8051	0.00268	4094	1498	746
	60	185	1850	196	3.65	7552	0.00264	3894	1476	745
	60	185	3700	165	3.07	6722	0.00250	3544	1444	739
	60	185	5550	142	2.65	6124	0.00237	3262	1420	731
	60	185	7400	129	2.40	5674	0.00232	3047	1388	729
	60	185	9250	117	2.19	5330	0.00225	2872	1365	727
	60	185	18,500	89	1.66	4332	0.00210	2312	1257	719

[a] Nuclear bimodal trip times are given in hours.

Table 4-13. (concluded)

Technology	Trip Time (days)	Initial Perigee (km)	Initial Apogee (km)	Thrust (N)	Propul-sive Power (kW)	Initial Mass (kg)	Initial Force/ Weight	ΔV (m/s)	Payload (kg)	Average I_{sp} (s)
Solar Thermal	30	185	185	44.5		8618	0.00053	4760	1534	866
	30	185	500	43.3		8392	0.00053	4516	1593	865
	30	185	700	42.5		8244	0.00053	4396	1611	865
	30	185	1000	41.6		8051	0.00053	4250	1627	865
	30	185	1850	39		7552	0.00053	3951	1622	864
	30	185	3700	34.7		6722	0.00053	3535	1545	862
	30	185	5550	31.6		6124	0.00053	3249	1462	861
	30	185	7400	29.3		5674	0.00053	3035	1387	860
	30	185	9250	27.5		5330	0.00053	2864	1325	859
	30	185	18500	22.4		4332	0.00053	2284	1132	856
	60	185	185	26.7		8618	0.00032	4616	1776	859
	60	185	500	26		8392	0.00032	4419	1808	858
	60	185	700	25.5		8244	0.00032	4319	1814	858
	60	185	1000	24.9		8051	0.00032	4192	1817	858
	60	185	1850	23.4		7552	0.00032	3919	1790	857
	60	185	3700	20.8		6722	0.00032	3524	1689	855
	60	185	5550	19		6124	0.00032	3246	1592	854
	60	185	7400	17.6		5674	0.00032	3032	1508	853
	60	185	9250	16.5		5330	0.00032	2862	1439	852
	60	185	18500	13.4		4332	0.00032	2284	1228	849

Table 4-14. Technology Performance Data for Titan IV Launch and Transfer to ORM 1 (GEO)

Technology	Trip Time (days)	Initial Perigee (km)	Initial Apogee (km)	Thrust (N)	Propulsive Power (kW)	Initial Mass (kg)	Initial Force/ Weight	ΔV (m/s)	Payload (kg)	Average I_{sp} (s)
Nuclear Bimodal	36.49[a]	185	185	2200		21637	0.0104	4555	4932	820.0
	34.32[a]	185	500	2200		20797	0.0108	4471	4790	820.0
	30.76[a]	185	700	2200		20310	0.0111	4435	4690	820.0
	27.56[a]	185	1000	2200		19618	0.0114	4378	4550	820.0
	25.07[a]	185	1850	2200		17928	0.0125	4142	4293	820.0
	22.38[a]	185	9250	2200		10988	0.0204	2902	2966	820.0
	31.22[a]	700	700	2200		19062	0.0118	4328	4432	820.0
Solar Bimodal	30	185	185	1217	22.69	21637	0.00574	4323	3255	751
	30	185	500	1135	21.17	20797	0.00557	4235	3297	751
	30	185	700	1096	20.43	20310	0.00551	4184	3305	751
	30	185	1000	1033	19.25	19618	0.00537	4105	3324	749
	30	185	1850	929	17.32	17928	0.00529	3910	3248	748
	30	185	3700	745	13.90	15225	0.00500	3553	3118	741
	30	185	5550	621	11.58	13365	0.00474	3263	2983	731
	30	185	7400	546	10.18	12014	0.00464	3049	2841	729
	30	185	9250	486	9.06	10988	0.00451	2875	2726	728
	30	185	18,500	337	6.29	8210	0.00419	2318	2320	721
	60	185	185	603	11.24	21637	0.00284	4318	4574	750
	60	185	500	562	10.48	20797	0.00276	4230	4528	750
	60	185	700	548	10.21	20310	0.00275	4168	4479	748
	60	185	1000	516	9.62	19618	0.00268	4094	4424	746
	60	185	1850	465	8.66	17928	0.00264	3894	4244	745
	60	185	3700	373	6.95	15225	0.00250	3544	3918	739
	60	185	5550	311	5.79	13365	0.00237	3262	3651	731
	60	185	7400	273	5.09	12014	0.00232	3047	3428	729
	60	185	9250	242	4.51	10988	0.00225	2872	3252	727
	60	185	18,500	169	3.14	8210	0.00210	2312	2696	719

[a]Nuclear bimodal trip times are given in hours.

Table 4-14. (concluded)

Technology	Trip Time (days)	Initial Perigee (km)	Initial Apogee (km)	Thrust (N)	Propul-sive Power (kW)	Initial Mass (kg)	Initial Force/ Weight	ΔV (m/s)	Payload (kg)	Average I_{sp} (s)
Solar Thermal	30	185	185	89.0		21637	0.00042	5029	4901	874
	30	185	500	85.5		20797	0.00042	4708	5055	874
	30	185	700	83.5		20310	0.00042	4551	5103	873
	30	185	1000	80.7		19618	0.00042	4365	5118	873
	30	185	1850	73.7		17928	0.00042	4008	50002	872
	30	185	3700	62.6		15225	0.00042	3554	4576	870
	30	185	5550	55		13365	0.00042	3259	4192	868
	30	185	7400	49.4		12014	0.00042	3040	3875	867
	30	185	9250	45.2		10988	0.00042	2868	3617	866
	30	185	18500	33.8		8210	0.00042	2285	2875	862
	60	185	185	66.7		21637	0.00031	4620	5570	871
	60	185	500	64.1		20797	0.00031	4421	5568	870
	60	185	700	62.6		20310	0.00031	4321	5543	870
	60	185	1000	60.5		19618	0.00031	4194	5481	869
	60	185	1850	55.3		17928	0.00031	3920	5251	868
	60	185	3700	46.9		15225	0.00031	3525	4739	866
	60	185	5550	41.2		13365	0.00031	3247	4322	865
	60	185	7400	37		12014	0.00031	3032	3992	863
	60	185	9250	33.9		10988	0.00031	2862	3723	862
	60	185	18500	25.3		8210	0.00031	2284	2957	858

also given as a function of drop-off conditions. Thrust for the solar bimodal and solar thermal transfer vehicles can be varied by changing the collector area and, in the solar bimodal case, the mass of the thermal exchange system (TES).

Maximum Payload Mass. For the nuclear bimodal systems, the tables show maximum payload corresponds to minimum drop-off altitude, viz., 185 × 185 km. This is expected because it takes maximum advantage of the high nuclear bimodal I_{sp}. The same is not always true for the solar bimodal or solar thermal systems, where maximum payload corresponds to a higher drop-off altitude for the 30-day transfers on the Atlas, Delta, and Titan launch vehicles and for the 60-day transfer on the Atlas vehicle.

In the solar bimodal case, low drop-off altitudes result in extensive eclipsing early in the transfer. To meet the trip time constraint, the initial poor performance resulting from the eclipses must be offset by a higher-thrust (heavier) propulsion system than would otherwise be required. (Remember that the solar bimodal system relies on storing thermal energy in the graphite absorber.) In the solar thermal case, the higher drop-off altitudes permit fewer and shorter perigee burns, thus reducing the magnitude of the thrust gravity losses.

Initial Thrust-to-Weight Ratio. For the low-thrust electric transfer vehicles, thrust is also variable. In the nuclear electric case, nuclear electrical power and thruster size can be varied. In the solar electric case, power is varied by varying the area of the photovoltaic arrays, the number of thrusters, and the size of the thruster PPU.

4.4 OECS COST/ENGINEERING MODEL (OCEM)

We have repeatedly emphasized the need for a common analysis methodology for all OECS technologies. The spreadsheet-based OECS Cost/Engineering Model (OCEM) is a critical element of that methodology. The model provides a consistent set of algorithms to select or size components and subsystems of the satellites and upper stages and cost them. OCEM is based on satellite and launch-vehicle sizing algorithms developed by Aerospace Corporation's Vehicle Design and Manufacturing Department.

4.4.1 Satellite Sizing Algorithms

The satellite sizing algorithms have been developed to perform preliminary design of satellite buses, given the requirements of the payload. These requirements include payload mass, payload electrical power, final orbit, pointing accuracy, etc. The algorithms allow sizing of eight subsystems: payload; propulsion; attitude determination and control (ADACS); telemetry, tracking, and command (TT&C); command and data handling (C&DH); thermal; power; and structure. The sizing of each subsystem is based on one of four methods:

1. Fixed weight and power
   - Payload
2. Database lookup
   - ADACS
   - TT&C
   - C&DH
3. Analytical relationships
   - Propulsion
   - Power
4. Historical satellite properties
   - Structure
   - Thermal control

Once the payload mass and power are input into the algorithms, the other seven subsystems are sized using the appropriate methods. For example, when database lookup is applied to the ADACS subsystem, an actual star sensor is chosen from a database of ADACS components. The mass, power, and model number of the star sensor are incorporated into the sizing algorithms. When sizing power system batteries, the battery mass and recharge power requirements are calculated with equations based on battery capacity, power density, and depth of discharge. In the case of satellite structure and thermal subsystems, sizing is best scaled with data from historical satellites. Subsystem size depends upon the properties of other subsystems. For example, structure weight is calculated as a percentage of the satellite dry mass, with the percentage based upon the structural materials. Table 4-15 presents the assumptions used in sizing each subsystem. A more detailed discussion of assumptions, including technology-specific assumptions, appears in Appendix B.

Table 4-15. Satellite Design Assumptions

Satellite Subsystem	OCEM Assumptions
ADACS	0.07° attitude knowledge 0.01° pointing accuracy
TT&C	SGLS downlink
C&DH	Integrated satellite processor (performs ADACS processing also) No data storage Mil-Std-1553B data bus
Thermal	Passive thermal control system (radiators, heat pipes, etc.)
Structure	Aluminum with selective use of composites
Growth Margin	15% for satellite bus

These sizing methods result in an interdependence of the subsystems. As an example, the structure subsystem is dependent on the dry mass of the entire satellite. Therefore, increasing the mass of any of the other subsystems will also increase the mass of the structure subsystem. Similarly, the thermal subsystem is dependent on the power consumed by the satellite. Therefore, increasing the power of any of the other subsystems will increase the mass and power of the thermal subsystem. Figure 4-6 shows the interdependency of the eight subsystems.

Once sized, the subsystem masses are summed to produce satellite dry mass. Propellants are added to the satellite dry mass to produce a satellite wet mass, which is then added to the launch vehicle adapter mass to produce the satellite launch mass. This launch mass is used to choose the appropriate launch vehicle. For the OECS, the launch vehicles of interest are the Delta II, the Atlas IIAS, and the Titan IV, which are shown to scale in Figure 4-7. Selected launch capabilities of these launchers are given in Table 4-16.

4.4.2 Launch-Vehicle Sizing Algorithms

The launch-vehicle sizing algorithms have been developed to perform the preliminary design of launch vehicles. In the OECS, they are used to size the cryogenic stages and the hydrogen tanks for the hydrogen arcjet, solar thermal, solar bimodal, and nuclear bimodal technologies. The algorithms consist of weight estimating relationships (WERs) based on the mass properties of historical US launch systems. Once the type and amount of propellant for each stage is input in the model, each stage of the launch vehicle is sized. The algorithms are broken down into four main subsystems: structure, consisting of tank, adapters, and thrust structure; propulsion, consisting of engines and plumbing; avionics; and thermal control.

4.4.3 Costing

Once a satellite/upper stage combination is sized, it has to be costed. The cost estimating relationships (CERs) used for the study were developed primarily by Rocketdyne and reviewed by Aerospace Corporation. Individual production and development CERs are determined at the component level for the power and propulsion subsystem and at the subsystem level for all other subsystems. Rocketdyne initially supplied one set of CERs for each technology, resulting in thirteen sets of CERs. These thirteen sets were then incorporated into one master set, allowing the costing of every possible component in all OECS configurations. This master set is linked directly to the OECS sizing algorithms in OCEM. A typical CER is of the form:

$$aX^b + c$$

where the coefficients a, b, and c are dependent on the component or subsystem, and X is the independent variable in the sizing algorithms. For example, the CER for the fuel tank is dependent on the volume of the fuel tank, thus:

$$Cost_{fueltank} = a(Volume_{fueltank})^b + c$$

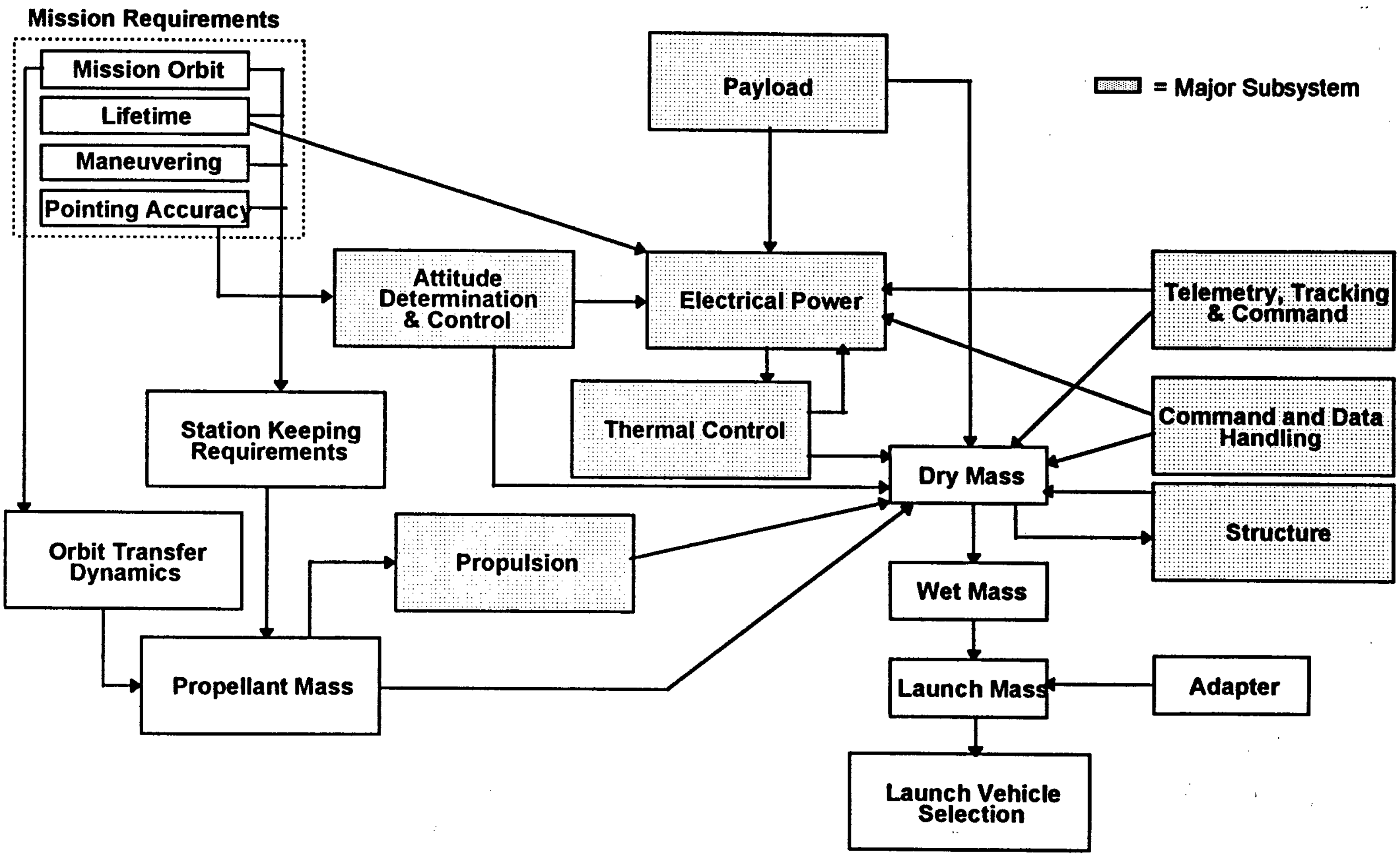
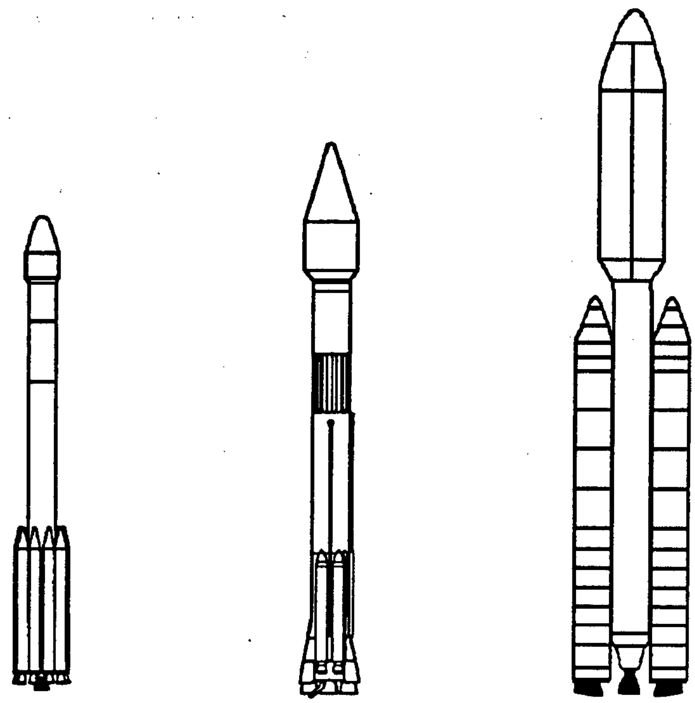


Figure 4-6. Interdependency of eight satellite subsystems.



Delta II 7920/7925 Atlas IIAS Titan IV SRMU

Figure 4-7. Launch vehicles used for the OECS analysis.

Table 4-16. Capabilities of the Principal Launch Vehicles Considered in the OECS

	Delta II 7920/7925	Atlas IIAS	Titan IV SRMU
100 nm, 28.5°	5039 kg (11,110 lb)	8639 kg (19,050 lb)	22,298 kg (49,167 lb)
200 nm, 28.5°	4787 kg (10,556 lb)	7868 kg (17,349 lb)	17,883 kg (39,432 lb)
100 x 450 nm, 98.7°	3447 kg (7600 lb)	6735 kg (14,850 lb)	16,442 kg (36,254)
450 nm, 98.7°	3175 kg (7000 lb)	5805 kg (12,800 lb)	—
GPS Transfer Orbit	1898 kg (4184 lb)	3855 kg (8500 lb)	5875 kg (12,954 lb)
GEO Transfer Orbit	1819 kg (4010 lb)	3696 kg (8150 lb)	—
GEO	—	—	5215 kg (11,500 lb)[a]

[a]Titan IV SRMU to GEO requires Titan Centaur upper stage.

The total system cost is calculated using the following equation:

$$Total\ Cost = \$A + \$D + \$FD + \$F + n(\$P + \$LV)$$

where

$\$A$ = technology acquisition cost
$\$D$ = satellite development cost
$\$FD$ = flight demo cost
$\$F$ = facilities cost
n = number of satellites
$\$P$ = average satellite production cost
$\$LV$ = launch vehicle cost

The cost from the above equation is the cost used in the cost-effectiveness analysis. A detailed discussion of the cost model is found in Chapter 6. The CERs are in Appendix F, a separate appendix in electronic format.

4.4.4 Applications

4.4.4.1 COST-EFFECTIVENESS ANALYSIS

The OCEM had to be automated for batch processing to handle the thousands of satellite and upper stage configurations required by the OECS. The original sizing algorithms were modified by removing the power and propulsion sections and replacing them with algorithms for the OECS technologies: three different power subsystems (photovoltaic, nuclear, and solar bimodal) and thirteen different propulsion subsystems (chemical, cryogenic, solar and nuclear electric [hydrogen arcjet, ammonia arcjet, xenon SPT, and xenon ion engines], solar thermal, solar bimodal, and nuclear bimodal). Each subsystem is sized concurrently; the input parameters determine which power and propulsion subsystems are incorporated. For example, if solar thermal technology is chosen for the upper stage, OCEM incorporates the photovoltaic power subsystem and the solar thermal propulsion system and then sizes and costs the combination of satellite and upper stage. The sizing/costing process is shown in Figures 4-8 and 4-9.

Presently, OCEM can size and cost a satellite/upper stage for any combination of the three power subsystem technologies, the thirteen propulsion technologies, and all OECS orbits. Additional technologies or orbits can easily be incorporated into the model in the future.

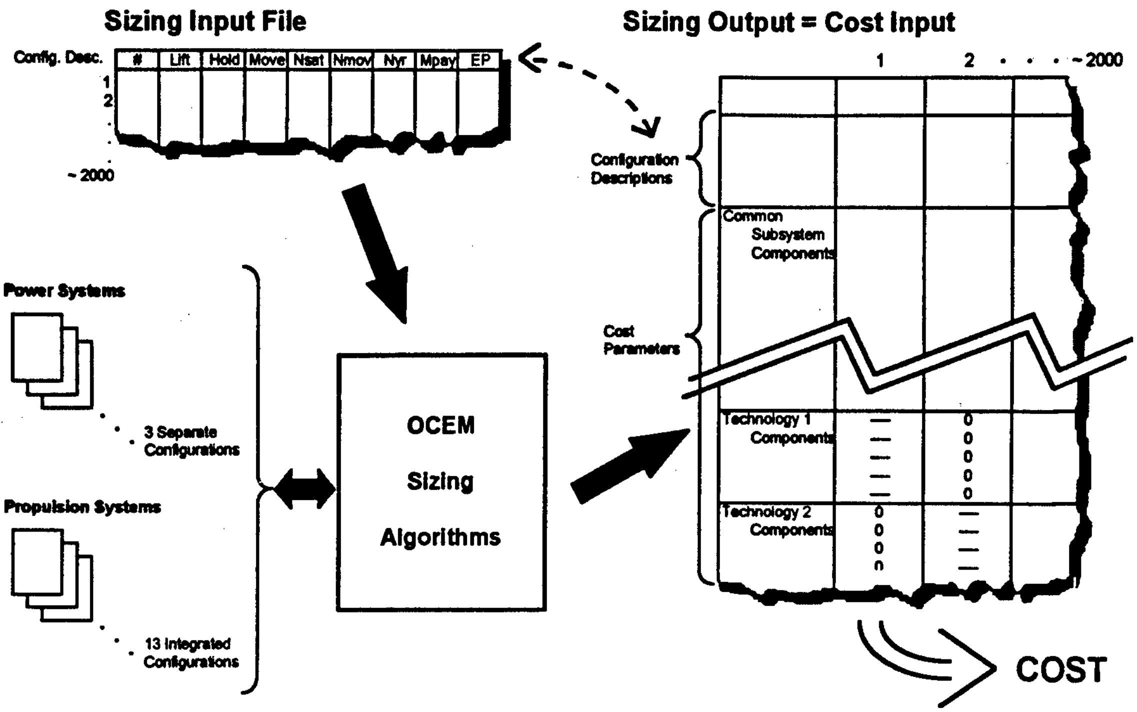


Figure 4-8. Sizing satellites for the cost-effectiveness analysis.

Typical single-case input in OCEM consists of the following:

- Analysis type (cost-effectiveness, effectiveness)
- Lift technology
- Hold technology
- Move technology
- Power technology
- ORM number
- Cost-effectiveness analysis parameters
  - Number of satellites
  - Number of moves
  - Satellite MMD
  - Payload mass
  - Payload power

OCEM can generate an input file consisting of a large number of cases. Once the input file is generated, OCEM sizes and costs the satellite/upper stage combination for each case. The cost information is inserted into the output file, which is used as input to the cost-effectiveness analysis.

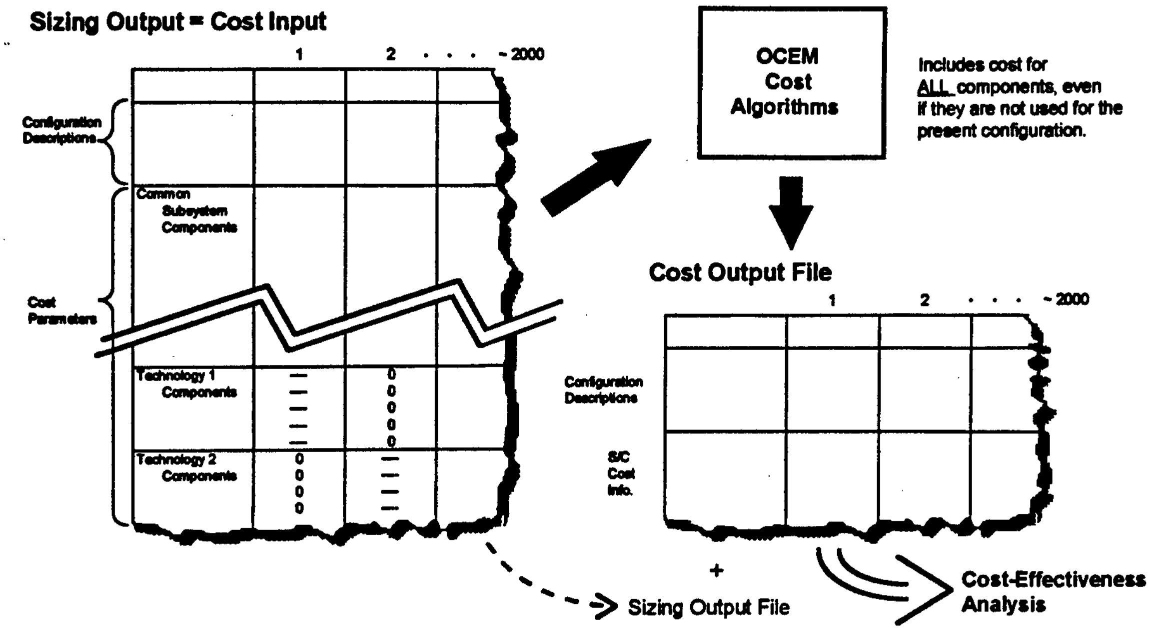


Figure 4-9. Costing satellites for the cost-effectiveness analysis.

4.4.4.2 EFFECTIVENESS ANALYSIS

OCEM effectiveness outputs are related to many of the measures of performance (MOPs) discussed in detail in Chapter 5. The effectiveness analyses are comparisons of the innovative technologies with existing satellite technology (i.e., bipropellant transfer, photovoltaic power). To make these comparisons, nominal payloads must be defined to serve as points of reference. Section 5.4.2 describes these nominal payloads in terms of payload mass, payload electrical power, and ΔV. The nominal payloads are designed to leave zero launch margin for a launch vehicle equipped with the baseline technology. They are used in all phases of the effectiveness analyses. The effectiveness analysis process is shown in Figure 4-10.

The maximum additional performance of the innovative technologies (MOEs 2-1 through 2-3) is calculated by holding two of the quantities *mass*, *power*, and ΔV at their nominal reference value and maximizing the value of the third quantity. In each case, the launch vehicle margin is zeroed by iterating on the quantity being maximized.

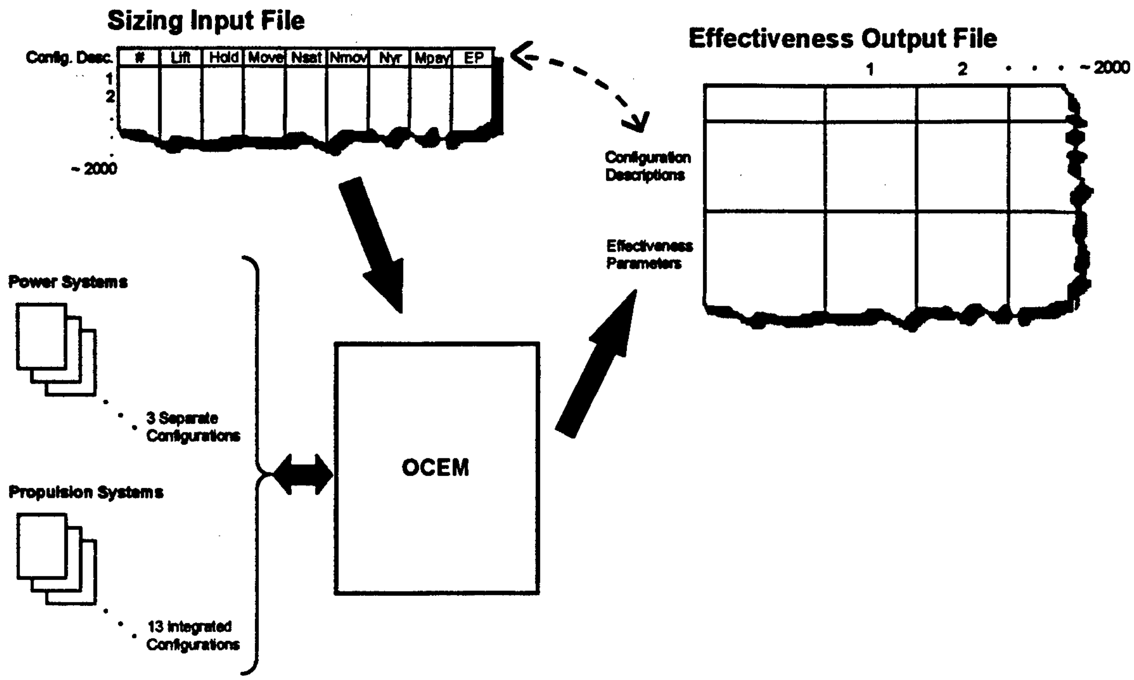


Figure 4-10. Process used for the effectiveness analysis.

Step-down (MOP 3-1) is the process of moving all or part of a payload from a larger to a smaller launch vehicle. OECS step-down is based on the reference payloads mentioned above. Holding the reference electrical power and ΔV constant, the OCEM determines the percentage of reference payload mass that can be launched with innovative technologies on step-down launch vehicles. OCEM zeroes out the performance of the step-down launch vehicle by varying step-down payload mass.

To determine step-down to a hypothetical launch vehicle (MOP 3-2), OCEM takes each innovative technology and calculates the combined upper stage/satellite mass delivered to drop-off to effect 100% step-down of the previously discussed reference payloads. This number can be compared with the corresponding baseline capabilities of the launch vehicle.

OCEM estimates the fairing volumes (MOP 1-3) required by each innovative technology. Volumes are based on payload and bus masses (assuming a standard density); individual propulsion and power considerations; and technology considerations such as packaged solar cell or solar collector volumes, nuclear reactor length, and fuel tank volumes. Details of these assumptions are found in Appendix D.

5. EFFECTIVENESS ANALYSIS OF ALTERNATIVES

The OECS analysis is structured like a cost and operational effectiveness analysis (COEA). It is based on a hierarchy of functional objectives (FOs), measures of effectiveness (MOEs), and measures of performance (MOPs). The FOs indicate in broad terms what the alternatives—in our case the technology combinations of Table 4-8—are expected to do. The MOEs provide yardsticks for measuring the general success of the alternatives, while the MOPs are the specific measures of performance that support the MOEs. In this chapter, we discuss the four FOs for lift, hold, move, and power, their MOEs and supporting MOPs, and the analysis results.

Effectiveness analyses typically establish performance norms that are useful in structuring and interpreting performance and cost results. Norms are either baselines against which alternatives can be compared or defined levels of performance that lead to consistent comparisons. Baselines are usually designed to correspond to current capability, while levels of performance are associated with defined, meaningful tasks.

The OECS has to determine four sets of norms for each ORM and launch vehicle:

- How much payload mass, electrical power, and on-orbit ΔV can the baseline technologies place on-orbit?
- How many satellites must be bought for each technology combination to achieve various levels of constellation availability?
- How much of the maximum payloads launched with the baseline technologies can be launched on smaller launch vehicles with innovative upper stage technologies?
- What throw weights would future launch vehicles need if they employed innovative upper stage technologies and provided complete step-down?

The first set of norms are used to measure the general improvement in the performance of the innovative technologies over the baseline. The second set allows equal cost-effectiveness comparisons. The third speaks to potential savings from replacing currently required launch vehicles with smaller and less costly existing launch vehicles, and the fourth provides significant insight into the necessary performance of future launch vehicles given the innovative technologies are adopted.

5.1 FUNCTIONAL OBJECTIVES (FOs)

5.1.1 FO 1: Lift Satellites to Initial Orbits

Historically, military space operations have relied on in-place constellations of satellites to meet mission needs. This will be true for the foreseeable future whether or not launch responsiveness improves. Thus, our ability to populate a constellation and maintain it for extended periods remains paramount. An indispensable element in this process is the upper stage (whether separate from or integrated with the satellite), which is used to move satellites from parking or transfer orbits into initial mission orbits. All the OECS lift technologies have a potential to provide this function.

5.1.2 FO 2: Hold Satellites in Operational Orbits

Orbital perturbations such as the nonsphericity of the earth, atmospheric drag, and third-body effects cause satellite orbital parameters to vary. In many cases, these changes must be corrected to maintain the satellite in an operational orbit. This correction process is known as *stationkeeping*. It is accomplished by thrusting with the satellite's on-board propulsion system. A satellite that has lost the ability to stationkeep for any reason will not maintain mission orbit tolerances, and its usefulness will be diminished or lost.

5.1.3 FO 3: Move Satellites to Other Orbits

There are many reasons why on-orbit maneuvering could be operationally desirable (see sidebar in Section 2.1.1). Some possibilities are: providing crisis coverage, disposing of satellites at end-of-life, redistributing functioning satellites after a satellite failure, replacing failed satellites with on-orbit spares, evading threats, moving to avoid orbit characterization, and reducing operational constellation size. While only a few of these reflect current operational practice, the need to move satellites is real. Increased maneuverability might permit new operational tactics or strategies.

5.1.4 FO 4: Power Satellites' Mission and Housekeeping Functions

A satellite's power subsystem must provide adequate electrical power for both mission and housekeeping functions throughout the satellite's life. All the OECS electrical power technologies can provide significant power.

5.2 MEASURES OF EFFECTIVENESS

Four general MOEs were chosen to measure how well each technology combination meets the functional objectives. They are:

- MOE 1: Constellation maintenance
- MOE 2: Improved on-orbit satellite capabilities
- MOE 3: Alternative launch vehicles
- MOE 4: Technology mission impact

The first three MOEs are predominantly quantitative. MOE 4 is qualitative. All MOEs seek to identify and compare each technology's functional utility. Figure 5-1 shows how the MOEs support the functional objectives.

The analyses for MOE 2 and MOE 3 generally follow one of two basic approaches: "bottom-up" or "top-down." Both approaches were discussed previously (Section 4.2.4). The bottom-up approach starts with a given launch vehicle and a given combination of lift, hold, move, and power technologies, then determines the payload (mass and power) and the

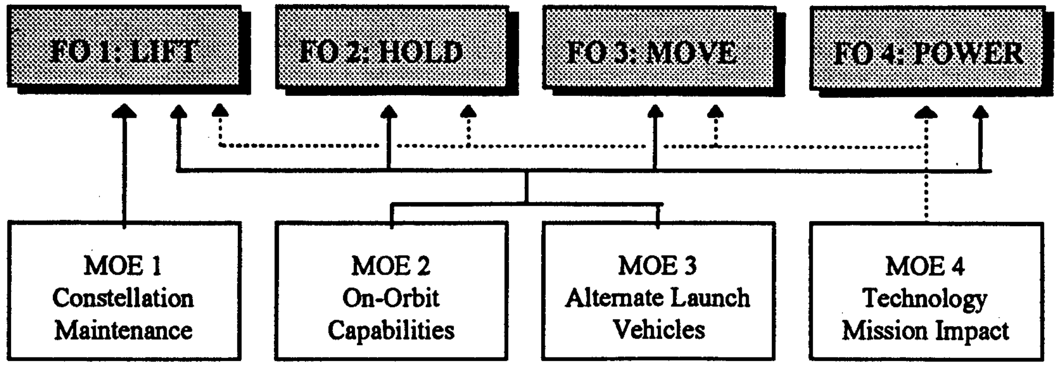


Figure 5-1. Relationship of functional objectives (FOs) and measures of effectiveness (MOEs).

satellite maneuver ΔV that can be placed in mission orbit. The top-down approach starts with a definition of a satellite payload (mass, power) and satellite maneuver ΔV, and for a given technology combination determines the launch vehicle and upper stage characteristics (including cost) needed to launch and place the satellite in mission orbit.

The OECS scenarios are defined by the ORMs, as described in Chapter 2. Technology combinations are evaluated with respect to their ability to support each ORM (see Section 4.3.8).

5.3 MOE 1: CONSTELLATION MAINTENANCE

5.3.1 Background and MOPs

The OECS must answer two quantitative questions about constellations: (1) Given a technology combination, an ORM, and a constellation, how well on average can the constellation be supported? and (2) How many satellites and launchers must be purchased to provide that support? The first question highlights the operational implications of the technologies' differing trip times (hence deployment time). The second is necessary to support the cost-effectiveness analysis.

In addition to these quantitative questions, there are two qualitative issues to be addressed: (1) What impact will each technology have on the launch vehicle? and (2) What are the possible technology-specific reliability issues?

These four questions are addressed, respectively, by four MOPs:

MOP 1-1: Availability of constellation (see discussion of concept in Section 4.3.1)
MOP 1-2: Acquisition of satellite and launch vehicle
MOP 1-3: Impact of lift technology on launch vehicle
MOP 1-4: Reliability of lift technology

MOP 1-1 determines the availability of a constellation. MOP 1-2 determines the average number of satellites and launch vehicles required to achieve this availability. MOP 1-1 and MOP 1-2 are determined by simulating the establishment and maintenance of the constellations for 15 years. MOP 1-3 examines possible physical conflicts of the technologies with current launch vehicle fairings. Potential conflicts are a function of satellite payload mass, payload electrical power, and on-orbit ΔV, because upper stage and satellite dimensions and masses are functions of these quantities. Evaluation of this MOP is done in OCEM (see Section 4.4). MOP 1-4 identifies potential reliability issues inherent in the technologies and how these issues can be modeled.

5.3.2 Methodology

The primary tool supporting MOP 1-1 and MOP 1-2 is the GAP_PLUS simulation model. This model is related in concept and capabilities to the Aerospace Corporation's Generalized Availability Program (GAP) and Starfleet Model, and AF Space Command's OSCARS Model. GAP_PLUS is a Monte Carlo simulation whose basic function is to simulate the establishment and maintenance of satellite constellations. The two principal outputs of the model are the constellation availability and the average number of satellites and launch vehicles required to support that availability.

GAP_PLUS is run in two phases. In the first phase, no constraints are placed on the number of satellites available for launch. The availability from this phase is the maximum availability that can be achieved for the given inputs. In the second phase, a desired availability of less than the maximum is specified. The model iterates on the delivery schedule of the satellites until the specified availability is achieved. Identifying equal availabilities in Phase 2 ensures that equal effectiveness will be costed in the cost-effectiveness analysis, because equal availabilities guarantee essentially equal constellation maintenance results (see sidebar on the next page).

GAP_PLUS inputs, with their relationship to previously discussed concepts in parentheses, are:

- Constellation size and constellation configuration (ORM, sparing policy)
- Deployment time, that is, the time from call-up to operational satellites (lift technology)
- Satellite reliability (satellite MMD)
- Launch vehicle reliability (launch vehicle and upper stage reliability)
- Minimum time between launches (launch rate constraints)
- Desired constellation availability (fixing the task to be accomplished)

OECS constellation lifetime is fixed at 15 years after establishment. Establishment consists of the first *n* launches, where *n* is the constellation size without active spares.

5.3.2.1 MOP 1-1: CONSTELLATION AVAILABILITY

Constellation availability is a generally accepted measure of how well a constellation is operationally maintained. In every GAP_PLUS simulation, the model determines the maximum possible constellation availability for the simulation inputs. Generally a higher availability is more desirable if everything else is equal.

5.3.2.2 MOP 1-2: NUMBER OF SATELLITES REQUIRED

We assume it is necessary to buy equal numbers of satellites, launch vehicles, and upper stages to establish and maintain a constellation. The number bought significantly impacts the cost of constellation maintenance and thus the cost effectiveness. Like availability, the number required is an output of the GAP_PLUS simulation. It is easy to predict trends in the variations of the number required. The number decreases with increasing launch vehicle reliability, increasing satellite lifetime (MMD), and increasing minimum time between launches. It increases with increasing constellation size and number of on-orbit spares.

Equal Effectiveness

Equal availabilities by definition guarantee equal probabilities of having an operational constellation. They do not guarantee equal numbers or lengths of intervals of nonavailability (down times). This is illustrated in this GAP_PLUS-generated table representing a four-satellite ORM 1 (GEO) constellation at 0.96 availability:

Deploy-ment (day)	No. of Down Times	Avg. Down Time (yr)	Avg. Max Down Time (yr)
60	3.50	0.18	0.31
90	0.76	0.75	1.47
120	0.80	0.69	1.49
360	0.92	0.63	1.16

In this case, the 60-day deployment time requires no spares to achieve the desired availability, while the other three deployment times require one active spare. The reader should not assume the trends in this example are universal. At a minimum, they vary with constellation size. Whether the differences shown are operationally meaningful is not a topic examined in the OECS.

5.3.3 Results

We use response surface methodology and GAP_PLUS results to generate one or more approximation equations for each ORM. These equations model how the maximum availability and number of satellites required vary with constellation size, deployment time, number of spares, MMD, launch rate, and launch vehicle reliability. There are seven pairs of equations representing ORM 1 (GEO), ORM 2a (MEO-GPS), ORM 3a (LEO-Polar), and ORM 4 (HEO). Appendix C shows the coefficients of these equations. The equations are applicable to the parameter ranges in Table 2-1.

These equations were used to generate plots showing lines of constant maximum availability and number of satellites required as functions of constellation size and deployment time. Figure 5-2 shows the results for ORM 1 (GEO) with no spares. Dashed lines on the plot represent maximum availability, and solid lines represent the number of satellites required to achieve this availability. This plot was made for 0.9 launch vehicle reliability, a satellite MMD of 8.5 years, and a minimum time between launches of four months. The lift technologies listed on the left of the figure indicate their characteristic OECS deployment times. Figure 5-3 shows similar results for one active spare.

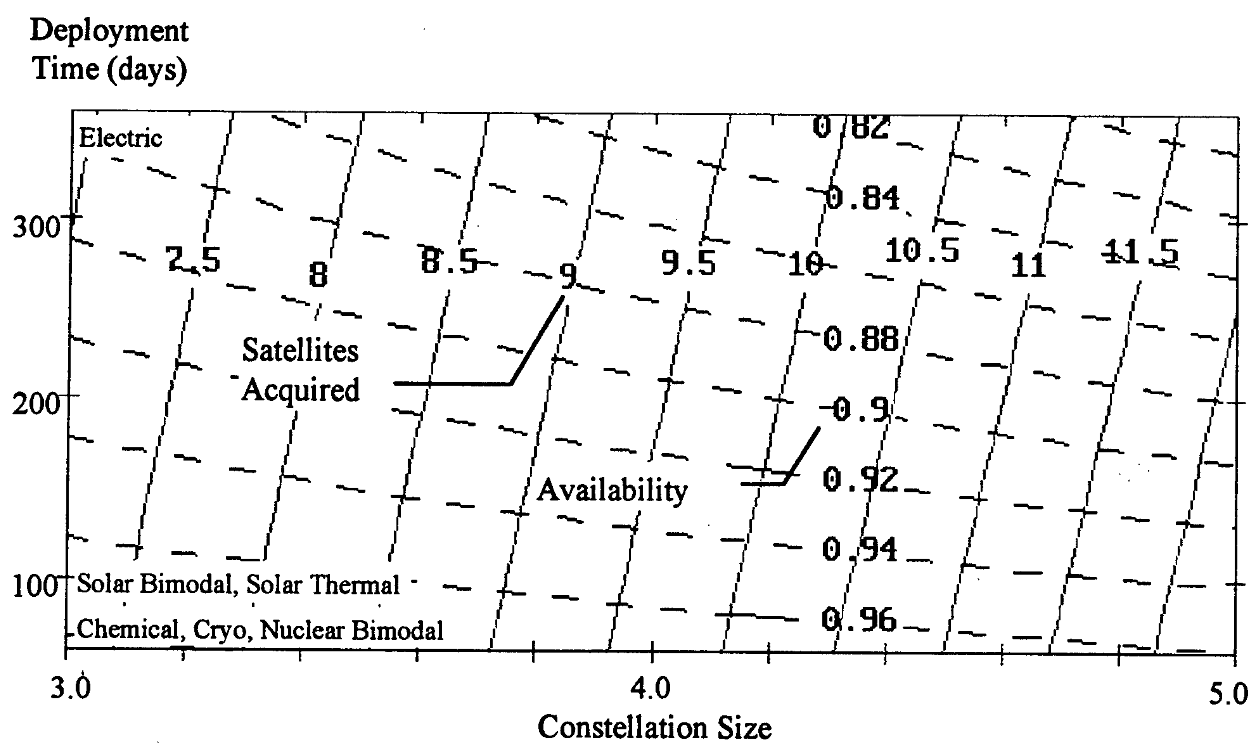


Figure 5-2. ORM 1 (GEO): availability and number of satellites required to maintain a constellation with zero spares.

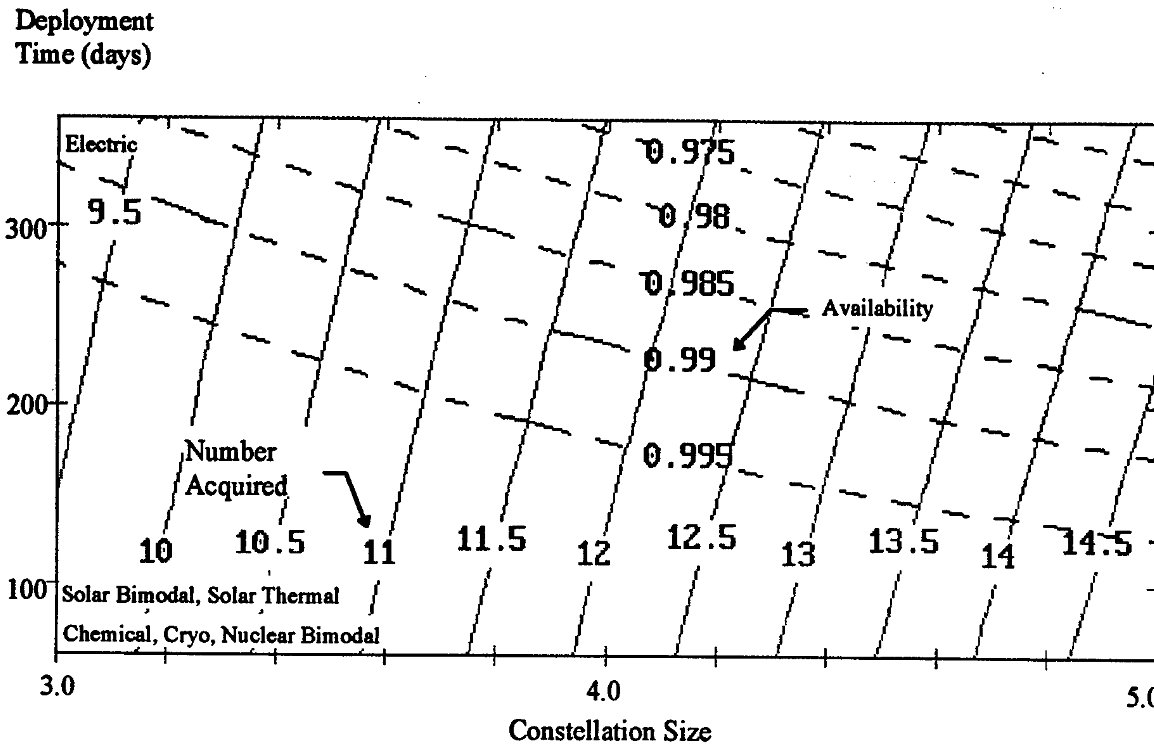


Figure 5-3. ORM 1 (GEO): availability and number of satellites required to maintain a constellation with one spare.

The importance of spares in achieving a high availability given long deployment times is obvious when Figure 5-2 is compared to Figure 5-3. Figure 5-2 shows that electric propulsion without a spare can achieve a maximum availability of less than 0.84 for a 4-satellite constellation. The availability for electric lift jumps to approximately 0.975 with an on-orbit operational spare. The penalty for using the spare for a constellation size of four, for example, is that the number of satellites required jumps from roughly 9 to 11.5. These additional satellites may be cost effective *if* the use of electric propulsion permits the use of a smaller, less expensive launch vehicle (launch vehicle step-down).

The previous discussion highlights one of the significant questions of the cost-effectiveness analysis: Can the same availability be achieved at less cost even though more satellites may need to be purchased? The answer strongly depends upon what upper stage technology is used. The situation is clearly different for the thermal systems (nuclear bimodal, solar bimodal, and solar thermal). Nuclear bimodal deployment times, because of short trip times, are comparable to the baseline and cryogenic systems. The additional month of trip time needed by the solar bimodal and solar thermal systems results in a small penalty in the availability and number of satellites required.

ORM 2 (GPS) represents a different situation because the constellation was designed with three active on-orbit spares. We see from Figure 5-4 that high availabilities are still achieved with three active spares when electric propulsion is used, despite the long deployment time.

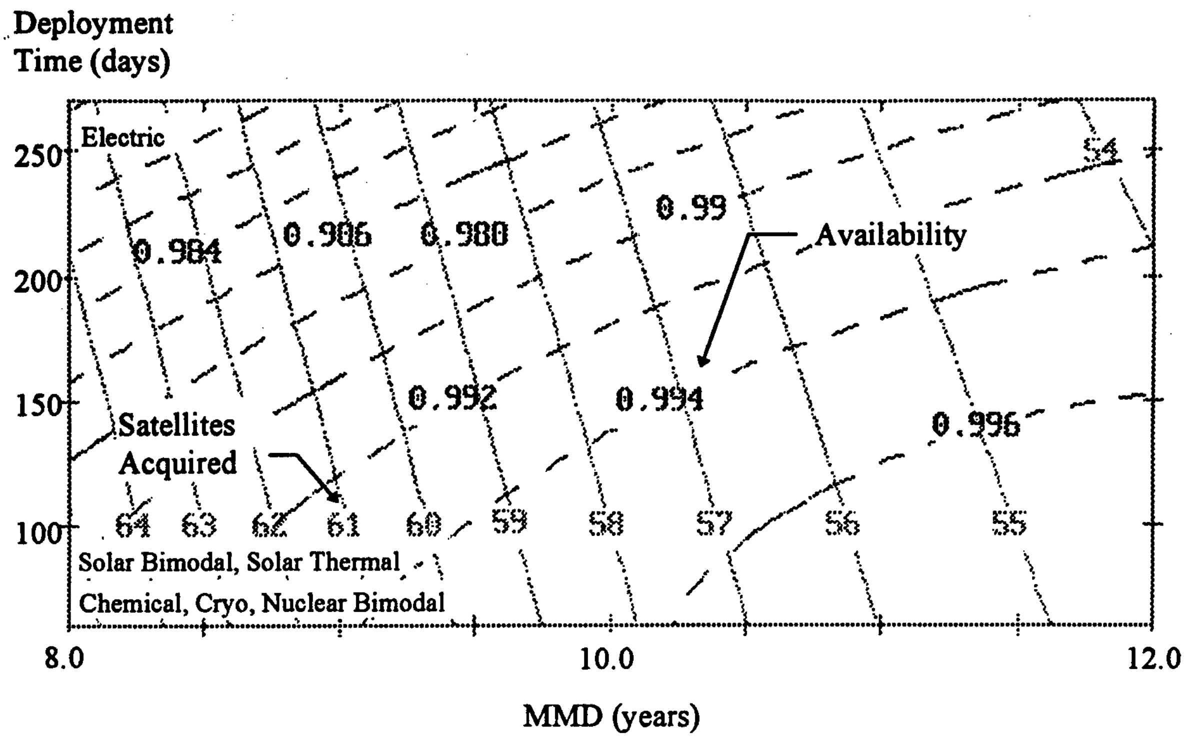


Figure 5-4. ORM 2a (GPS): availability and number of satellites required to maintain a 21-satellite constellation with 3 spares.

5.3.3.1 MOP 1-3: IMPACT OF LIFT TECHNOLOGY ON THE LAUNCH VEHICLE

A big concern with the innovative technologies is fitting them into existing launch vehicle fairings. This is especially a problem for the thermal technologies that use H_2 as a propellant. While new, larger fairings potentially could be crafted, their development would increase costs and possibly impact launch vehicle dynamics—raising more problems. To investigate this issue, we approximated the combined upper stage and satellite volumes (including unusable empty space) as they might be packaged in the fairings. These volumes were then compared to total launch-vehicle fairing volumes.

To estimate the combined satellite and upper stage volumes (with the exception of advanced cryo, which was treated as a fixed volume upper stage plus satellite volume), we treated the upper stage and satellite as consisting of three separate pieces:

- Upper stage (lift propulsion subsystem minus the propellant tank)
- Lift propellant tank
- Satellite (payload and bus minus the lift propulsion subsystem)

The sum of these three volumes is an approximation that, when compared with fairing volume, indicates the probable fit or lack of fit.

The upper stage volume (less propellant tank) is estimated individually for each technology. Propellant tank volumes are estimated by OCEM based upon propellant requirements. H_2 tanks, which are very large due to the low density of H_2, are considered to be cylinders with ellipsoidal end caps. The diameter of these cylinders is determined by the required propellant volume, the fairing diameter, and the lift technology. Other propellant tanks are treated simply as additive volumes. The satellite is considered to be a cylinder the diameter of the fairing. Its volume is based on a nominal average satellite density of 79 kg/m^3 (Larson and Wertz, p. 292).

If ratios of the calculated volumes to the fairing volumes are substantially less than one, there is a good possibility the technology can be accommodated by the current fairing. The large Titan fairing with the Centaur upper stage removed has an approximate volume of 238 m^3, while the large Atlas fairing and two-stage, 10-ft Delta fairings have volumes of approximately 68 m^3 and 31 m^3, respectively. Ratios substantially larger than one indicate a likely volume problem. (Appendix D has the packaging assumptions and volume computations.) Ratios near one are ambiguous given the uncertainty of the approximations.

Figures 5-5—5-7 summarize a sampling of volume ratios for ORM 1 (GEO), ORM 2 (MEO-GPS), and ORM 4 (HEO), respectively. Each point in the figures represents a satellite that maximizes on-orbit payload mass for the corresponding nominal payload electrical power, ΔV, and MMD values (these values are listed in Table 5-2). The ordinates show the ratio of estimated volume to fairing volume based on the largest available fairings. The abcissas show the payload mass ratios of innovative to baseline technologies. The higher the ratio, the greater the payload mass; that is, the better the lift performance.

The technology combinations are the best performing combinations of each lift technology. Because the lift technology dominates in determining total volume, these volumes are typical for any combination of hold and move technologies. There are potential techniques to reduce the required H_2 volume with only small losses in overall performance. Two are discussed in the following sidebar.

Volume Reduction

At least two techniques for reducing the overall volume of thermal systems have been suggested. Both call for paying a small performance penalty for a significant reduction in H_2 tank volume. The first relies initially on launching into a higher-energy, elliptical drop-off orbit rather than the circular one assumed in the OECS. This reduces upper stage ΔV requirements and, hence, the size of the upper stage H_2 tank. The second technique involves using higher-density, higher-thrust, lower-I_{sp} NH_3 as a propellant immediately after drop-off. The lower I_{sp} is more than compensated for by the higher thrust resulting from lower initial gravity losses when the upper stage and satellite are near the Earth. The combined volume of the two tanks is smaller than the original H_2 tank.

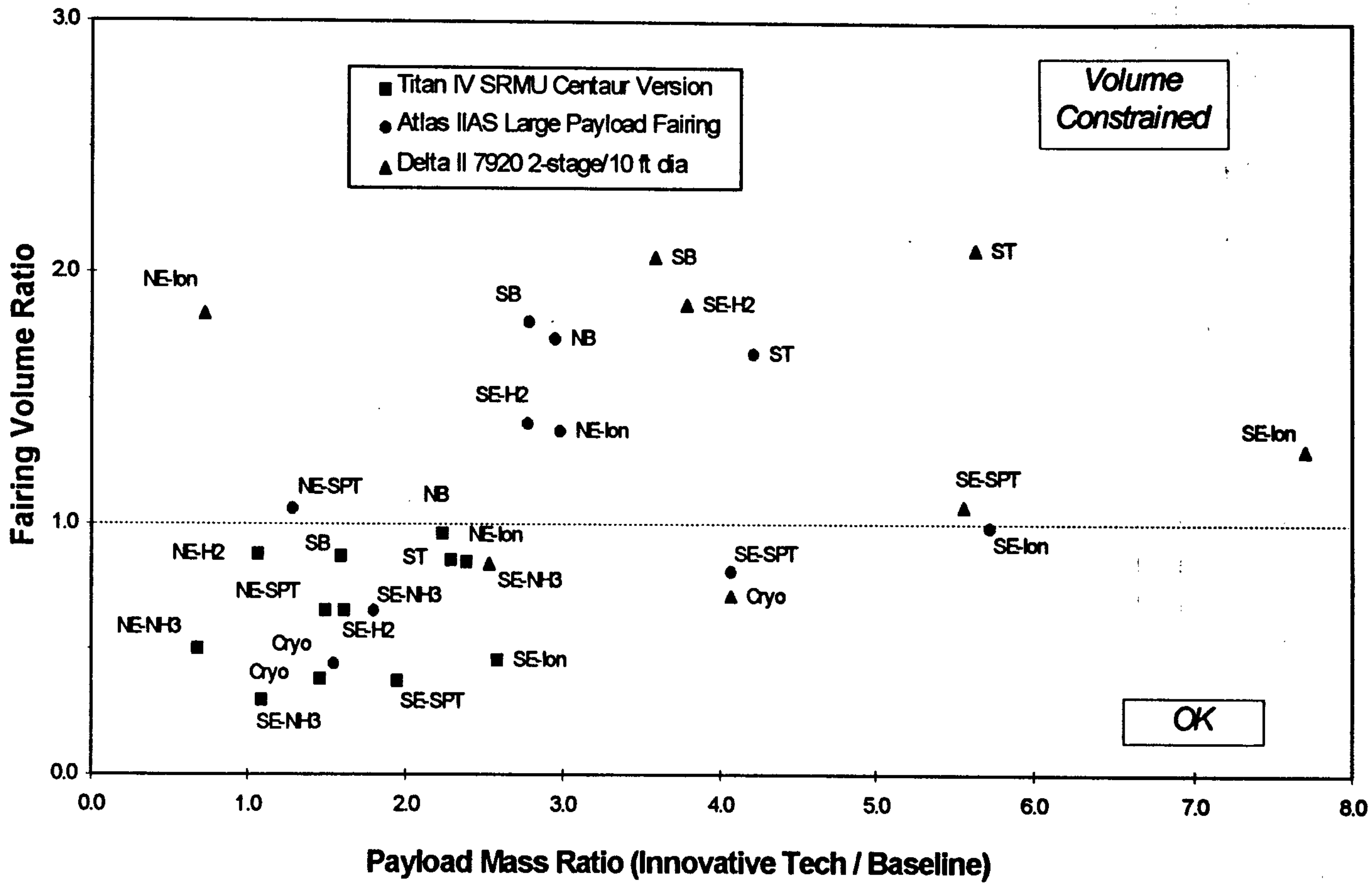


Figure 5-5. ORM 1 (GEO): approximate ratio of required fairing volume to largest current fairing volume of Delta, Atlas, and Titan.

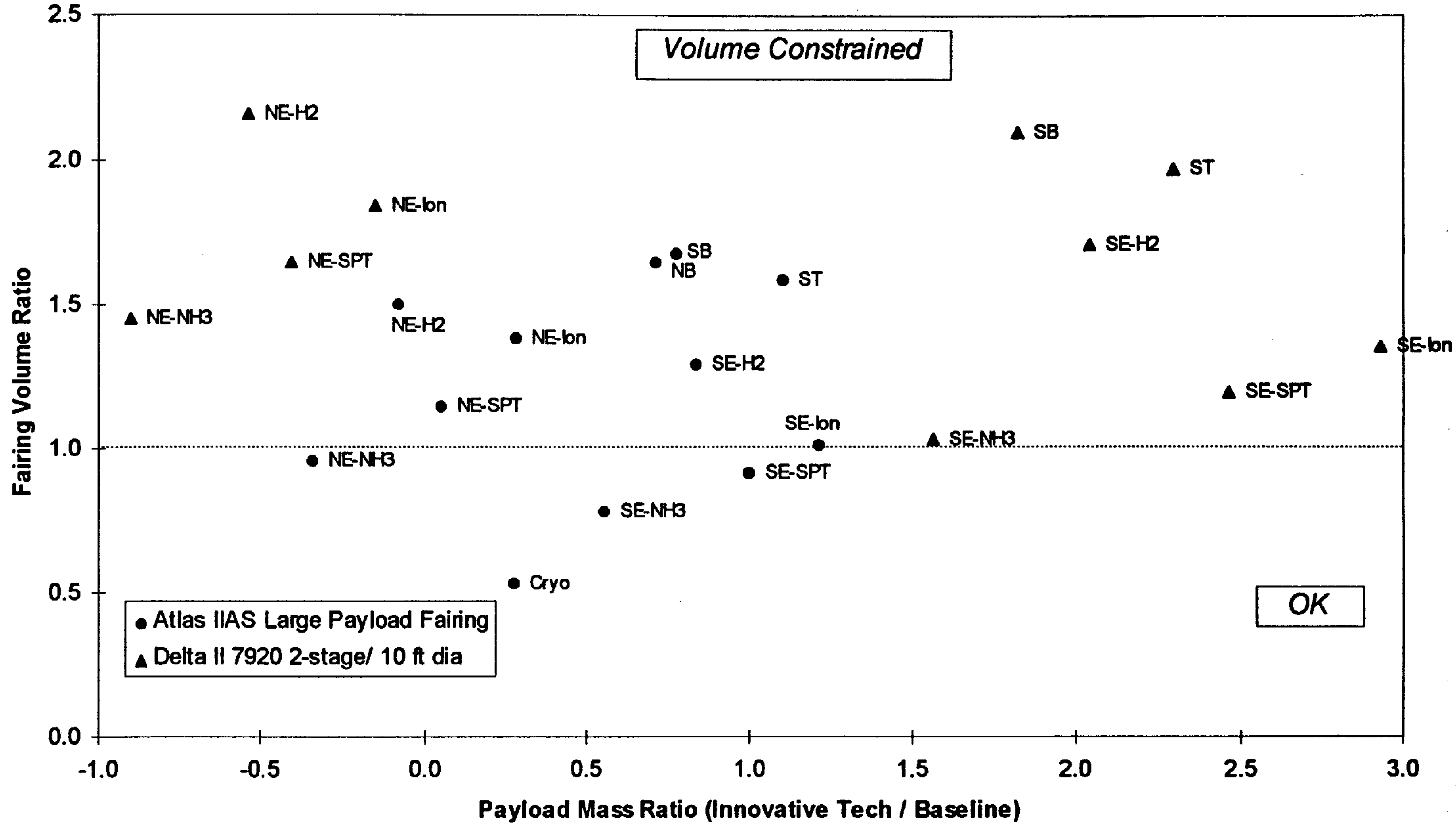


Figure 5-6. ORM 2a (GPS): approximate ratio of required fairing volume to largest current fairing volume of Delta, Atlas, and Titan.

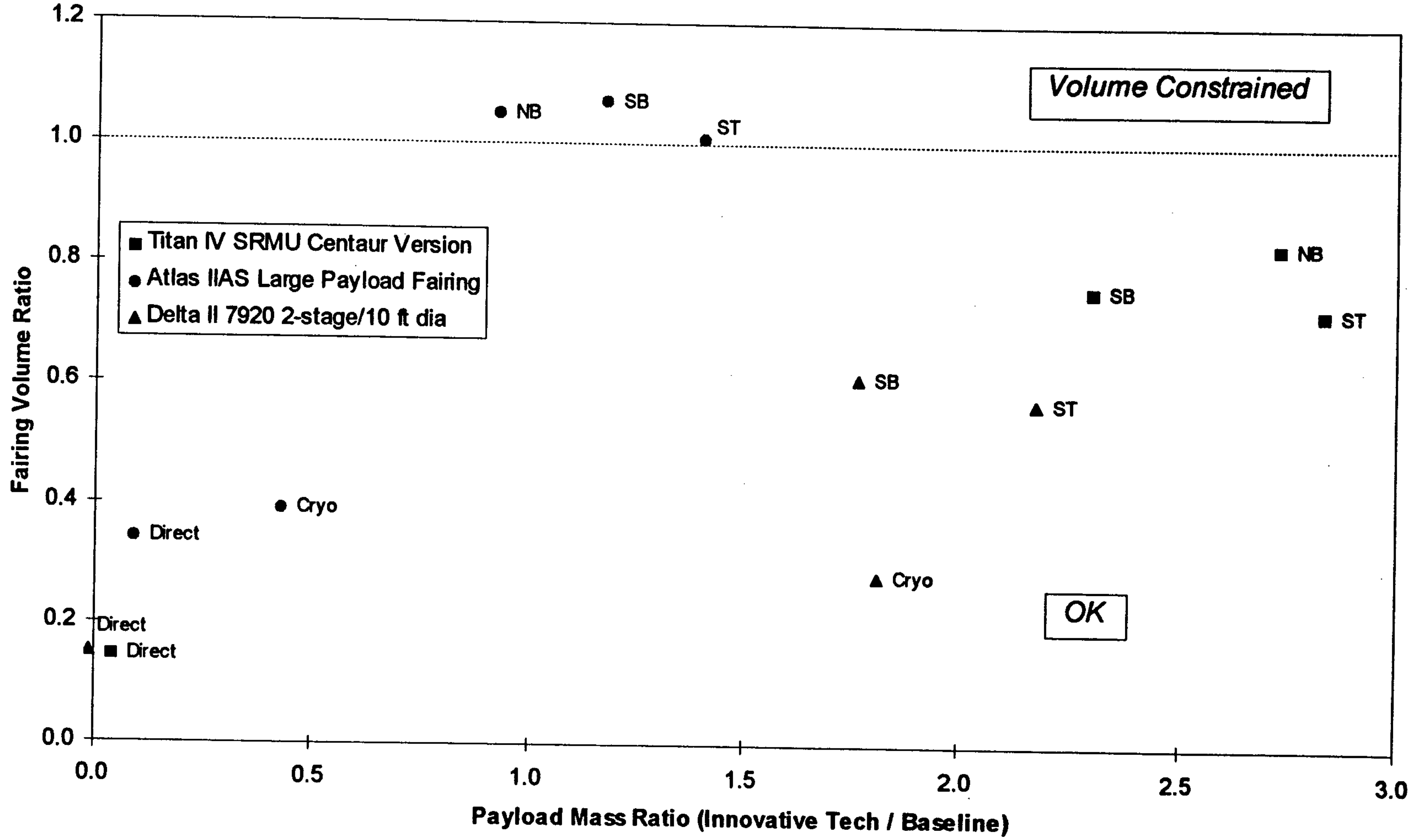


Figure 5-7. ORM 4 (HEO): approximate ratio of required fairing volume to largest current fairing volume of Delta, Atlas, and Titan.

5.3.3.2 MOP 1-4: RELIABILITY OF THE LIFT TECHNOLOGY

The OECS assumes all lift technologies have identical reliabilities. This study ground rule was adopted to avoid the almost impossible—and certainly contentious—assessment of the reliability of individual developmental technologies. In the GAP_PLUS analysis, lift reliability is subsumed in the launch vehicle reliability. The launch vehicle and the upper stage are assumed to contribute equally to the reliability. This assumption slightly favors technologies with longer lift times because all failures are modeled as occurring at launch.

MOP 1-4 is qualitative since no quantitative OECS reliability assessments have been made. It uses lift, hold, move, and electrical power engineering assessments to identify possible factors that may adversely affect reliability. Consideration is given to restart capability, propulsion system lifetime, propellant storage, natural environmental hazards, heat dissipation, etc. Many known reliability concerns have been considered in the technology designs. Table 5-1 lists concerns by technology and includes a summary of the consideration given them in the design process.

Table 5-1. Reliability Issues and Design Considerations

Reliability Issue	Design Consideration
Advanced Cryogenic	
Titan Centaur goes to GEO	Insulation and H_2 increased to counter boil-off
Nuclear Bimodal	
Reactor irradiation of payload and bus	Boom separating reactor from payload and bus; shielding
Solar Bimodal	
None identified	None identified
Solar Thermal	
Puncture of inflatable solar collectors by micrometeorites	Collectors disposed of immediately after orbital transfer
Nuclear Electric	
Arcjet thruster life	Multiple sets of arcjets used sequentially for lift
Ion thruster life	Multiple sets of ion thrusters for lift
Reactor irradiation of payload and bus	Boom separating reactor from payload and bus; shielding
Solar Electric	
Arcjet thruster life	Multiple sets of arcjets used sequentially for lift
Ion thruster life	Multiple sets of ion thrusters for lift
Photovoltaic cell radiation degradation	Photovoltaic arrays sized for all radiation degradation, including passage through Van Allen belts

5.4 MOE 2: IMPROVED ON-ORBIT CAPABILITIES

5.4.1 Background and MOPs

When flown on a given launch vehicle, innovative technology combinations are likely to provide more useable mass in mission orbit than the baseline technologies. This is due to the innovative technologies' higher I_{sp} values. The likelihood of more useable mass will be realized whenever the propulsion and power hardware masses plus the propellant masses of the innovative technologies are less than the corresponding baseline masses.

Additional on-orbit mass can be used to enhance satellite performance over the baseline capability. Enhancement could mean a simplified design, a functionally augmented or more robust payload or bus, more hold and move capability, higher electrical power, or a combination of these factors. In MOE 2, we are concerned with identifying the technology combinations that enhance capabilities as well as with quantifying the enhancement.

We have selected three enhancements that are well defined and easily quantified. Expressed as MOPs, they are:

- MOP 2-1: Additional on-orbit payload mass
- MOP 2-2: Additional on-orbit payload electrical power
- MOP 2-3: Additional on-orbit satellite ΔV

In each case, *additional* is measured relative to baseline technology performance represented by nominal payloads defined in Table 5-2. Each MOP is evaluated with respect to each technology combination and ORM for the Titan IV, Atlas IIAS, and Delta II.

A fourth MOP qualitatively examines the possible impact of the innovative technologies on their payloads.

- MOP 2-4: Propulsion and power subsystem impacts on the payload

Added functionality or reliability of the payload or bus as a result of additional payload mass, power, or maneuverability are extremely desirable areas to understand, but they are neither readily quantified nor easily interpreted. Exploring such implications is beyond the scope of the OECS (see sidebar).

An OECS Follow-On Study?

The *Solar Electric Propulsion Assessment* (Chan et al.) and the OECS were unable to evaluate the worth of additional on-orbit mass in an operational context. This omission is at best annoying and at worst a significant shortcoming to understanding the value of the innovative propulsion and power technologies. The value of extra mass on-orbit is intuitively obvious, but the flexibility it promises in satellite design and overall space architecture needs quantification. Such a study would be a worthwhile follow-on to the OECS.

5.4.3 Methodology

Before additional performance can be measured, we need to establish the performance of the baseline technologies. To do this, we establish representative payloads with zero unused lift capability (zero lift margin) for each ORM and launch vehicle when using the baseline technologies. Necessarily, each of these payloads is a single point on a payload envelope having zero lift margin. The envelopes result from trading payload mass, electrical power, and on-orbit ΔV as determined by maneuver ΔV and MMD (see Table 2.1 for stationkeeping ΔV/yr). Because of the zero lift margin constraint, choosing any two of mass, power, and ΔV determines the third.

Our approach is to select representative electrical power and ΔV values, then use OCEM (see Section 4.4) in bottom-up mode to determine the corresponding payload mass. Table 5-2 gives the resulting representative values for mass, power, MMD, and maneuver ΔV for each ORM and launch vehicle.

Table 5-2. Representative Payload Parameters of the Baseline Technology

ORM	Parameter	Launch Vehicle		
		Titan IV	Atlas IIAS	Delta II
ORM-1 (GEO)	EOL P/L Power (kW)	5.0	3.0	1.0
	ManeuverΔV (m/s)	51.6	51.6	25.8
	MMD (yr)	10.0	10.0	7.5
	Payload Mass (kg)	2254	380	172
ORM-2a (MEO-GPS)	EOL P/L Power (kW)	N/A	1.0	1.0
	Maneuver ΔV (m/s)	N/A	80.6	80.6
	MMD (yr)	N/A	10.0	10.0
	Payload Mass (kg)	N/A	1262	387
ORM-3a (LEO-Polar)	EOL P/L Power (kW)	N/A	2.0	1.3
	Maneuver ΔV (m/s)	N/A	120.0	20
	MMD (yr)	N/A	7.0	5.5
	Payload Mass (kg)	N/A	3843	1890
ORM-4 (HEO)	EOL P/L Power (kW)	2.8	2.8	1.0
	Maneuver ΔV (m/s)	190	190	95
	MMD (yr)	7.5	7.5	5.0
	Payload Mass (kg)	1665	852	365

5.4.3.1 MOP 2-1: ADDITIONAL ON-ORBIT PAYLOAD MASS

Similar methodologies were used for evaluating MOP 2-1 through MOP 2-3 with respect to the representative payloads. For MOP 2-1, OCEM is used in bottom-up mode to calculate the maximum payload mass that can be placed on orbit for each innovative technology combination and the appropriate representative satellite power and ΔV from Table 5-2. As for the baseline technology, this payload mass corresponds to a zero launch-vehicle margin. The difference between the baseline and innovative technology payload masses is the additional payload mass that can be placed on-orbit by the innovative technology.

Each technology combination from Table 4-8 is examined in turn. Additional payload mass may be negative in some instances, despite the higher I_{sp} values of the innovative technologies. This typically happens for the inherently heavy nuclear technologies, especially when implemented on smaller launch vehicles, or when the technology supports low ΔV requirements—essentially eliminating the possibility of significant potential advantage. Some innovative technologies may be too massive to fit on a given launch vehicle and still provide a useful capability. Payload fairing constraints, discussed earlier in conjunction with MOP 1-3, are not considered in the calculations.

5.4.3.2 MOP 2-2: ADDITIONAL ON-ORBIT PAYLOAD ELECTRICAL POWER

We use an analogous method to determine the additional on-orbit payload electrical power provided by the innovative technologies. In this case we fix the representative satellite mass and ΔV and use OCEM to maximize the available payload electrical power. The difference between the baseline satellite's payload electrical power and that of the innovative technology is the additional payload electrical power the innovative technology can place on-orbit. As with payload mass, the additional power may be negative, or the innovative technology may be too massive for the launch vehicle.

5.4.3.3 MOP 2-3: ADDITIONAL ON-ORBIT MANEUVER ΔV

The methodology for determining additional maneuver ΔV is analogous to that for determining additional mass and power. Generally the innovative technologies will provide additional ΔV if the corresponding additional payload mass is positive. Since the ΔV magnitude is usually very large compared to the baseline values, we will give less attention to additional ΔV than payload mass and electrical power.

5.4.3.4 INTEGRATING THE MOPS

The evaluation of these three MOPs defines three points in the three-dimensional space of Figure 5-8 whose axes are additional on-orbit payload mass, payload electrical power, and maneuver ΔV. The plane triangle constructed by connecting these three points has been observed to approximate the surface of maximum additional capabilities for the innovative technology in question. Thus any combination of additional mass, power, and maneuver ΔV lying beneath the triangle is attainable with the innovative technology. While such a plot provides significant information, it does not lend itself to comparisons with other innovative technologies. Therefore we have focused on comparisons of the maximum values, which are easy to comprehend and compare. These extremes are summarized ORM by ORM in the tables and figures on the following pages. In some cases we have also plotted additional payload mass as a function of additional payload electrical power for zero additional satellite ΔV.

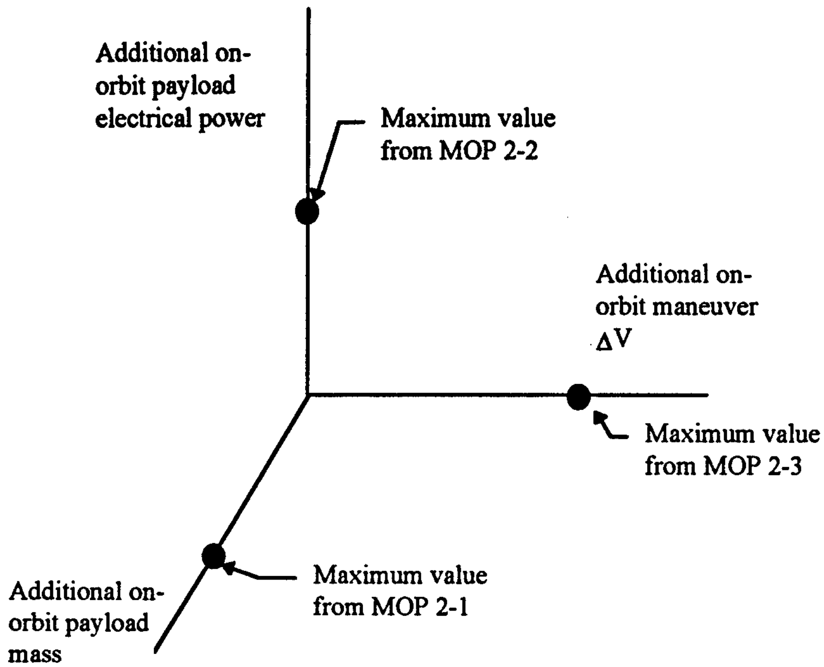


Figure 5-8. Trade space of the additional capabilities provided by the innovative technologies.

5.4.3 Results

5.4.3.1 ORM 1 (GEO): ADDITIONAL CAPABILITIES PROVIDED BY INNOVATIVE TECHNOLOGIES

ORM 1 (GEO) additional payload mass, payload electrical power, and maneuver ΔV capabilities are presented in Table 5-3 through Table 5-5 for Titan IV, Atlas IIAS, and Delta II, respectively. Estimated fairing volumes as percentages of the largest currently available fairings are also included. Each table's results are presented by technology combination. The first row of each table lists the baseline capabilities. The remaining rows give results for the other 27 GEO technology combinations from Table 4-8. For each lift technology, the first row corresponds to electric on-orbit move propulsion, the second to chemical. For the bimodal technologies, the third row corresponds to using the lift technology with NH_3 propellant for move. Since there is little difference in the results among the move technologies, only the results of electric move propulsion are reported throughout the remainder of this chapter.

Table 5-3. ORM 1 (GEO): Maximum Additional Payload Mass, Payload Power, and Maneuver ΔV That Can Be Attained With Innovative Technologies Launched on Titan IV

No.	Technology	P/L Mass (kg)	% Fairing	P/L Power (kW)	% Fairing	Maneuver Δv (m/s)	% Fairing
Titan IV Baseline Values							
1-1	Baseline Chemical	2254	27	5.0	27	52	27
Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV							
1-2	Chemical	–50	27	–0.3	27	–31	27
1-3	Advanced Cryo	+1030	38	+5.7	38	+1293	38
1-4		+964	38	+5.3	38	+498	38
1-5	Nuclear Bimodal	+2776	96	+19.8	96	+2050	96
1-6		+2658	96	+19.0	96	+808	96
1-7		+2752	96	+19.7	96	+1585	96
1-8	Solar Bimodal	+1335	87	+14.9	86	+920	87
1-9		+1219	87	+14.3	86	+346	87
1-10		+1307	87	+14.8	86	+674	87
1-11	Solar Thermal	+2909	85	+16.5	85	+2922	85
1-12		+2817	85	+16.0	85	1164	85
1-13	Nuclear NH_3	–718	50	a	a	a	a
1-14		–811	50	a	a	a	a
1-15	Nuclear H_2 Arcjet	+141	88	+2.1	88	+106	88
1-16		+42	88	+0.6	88	+13	88
1-17	Nuclear Xe SPT	+1371	65	+22.7	65	+2401	65
1-18		+1191	65	+19.6	65	+314	65
P1-19	Nuclear Xe Ion	+3118	84	+54.8	84	+9254	84
1-20		+2871	84	+50.2	84	+644	84
1-21	Solar NH_3 Arcjet	+184	30	+1.3	30	+231	30
1-22		+91	30	+0.6	30	+35	30
1-23	Solar H_2 Arcjet	+1095	65	+8.1	65	+705	64
1-24		+1011	65	+7.5	65	–52	63
1-25	Solar Xe SPT	+2147	37	+16.2	37	+3931	36
1-26		+1967	37	+14.8	37	+544	37
1-27	Solar Xe Ion	+3565	45	+27.5	45	+10824	44
1-28		+3318	45	+25.6	45	+763	45

[a]Launch vehicle capabilities exceeded: technology cannot support requirements. For *Max P/L Mass*, the payload power, moves, and MMD must equal that of the baseline; for *Max P/L Power* (or *Moves*), the payload mass, MMD and moves (or payload power) must equal that of the baseline (see discussion in MOP methodology).

Table 5-4. ORM 1 (GEO): Maximum Additional Payload Mass, Payload Power, and Maneuver ΔV That Can Be Attained With Innovative Technologies Launched on Atlas IIAS

No.	Technology	P/L Mass (kg)	% Fairing	P/L Power (kW)	% Fairing	Maneuver Δv (m/s)	% Fairing
Atlas IIAS Baseline Values							
1-1	Baseline Chemical	380	37	3.0	37	52	37
Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV							
1-2	Chemical	–19	37	–0.1	37	–31	37
1-3	Advanced Cryo	+208	44	+1.1	44	+764	44
1-4		+187	44	+1.0	44	+282	44
1-5	Nuclear Bimodal	+737	174	+5.0	174	+1279	174
1-6		+690	174	+4.7	174	+492	174
1-7		+728	174	+5.0	174	+981	174
1-8	Solar Bimodal	+677	181	+5.7	179	+1223	181
1-9		+632	181	+5.4	179	+469	181
1-10		+666	181	+5.6	179	+900	181
1-11	Solar Thermal	+1222	168	+6.7	168	+3300	168
1-12		+1187	168	+6.5	168	+1319	168
1-13	Nuclear NH_3 Arcjet	[a]	[a]	[a]	[a]	[a]	[a]
1-14		[a]	[a]	[a]	[a]	[a]	[a]
1-15	Nuclear H_2 Arcjet	[a]	[a]	[a]	[a]	[a]	[a]
1-16		[a]	[a]	[a]	[a]	[a]	[a]
1-17	Nuclear Xe SPT	+106	106	+1.5	106	+403	106
1-18		+28	106	+0.4	106	+16	106
1-19	Nuclear Xe Ion	+753	137	+11.7	137	+4798	135
1-20		+645	137	+9.9	137	+309	137
1-21	Solar NH_3 Arcjet	+302	65	+2.1	65	+908	65
1-22		+261	65	+1.8	65	+242	65
1-23	Solar H_2 Arcjet	+671	140	+4.8	140	+1620	140
1-24		+636	140	+4.5	140	+632	140
1-25	Solar Xe SPT	+1165	81	+8.5	81	+5055	79
1-26		+1087	81	+7.9	81	+714	81
1-27	Solar Xe Ion	+1793	99	+13.3	99	+12859	96
1-28		+1686	99	+12.4	99	+918	99

[a]Launch vehicle capabilities exceeded: technology cannot support requirements. For *Max P/L Mass*, the payload power, moves, and MMD must equal that of the baseline; for *Max P/L Power* (or *Moves*), the payload mass, MMD and moves (or payload power) must equal that of the baseline (see discussion in MOP methodology).

Table 5-5. ORM 1 (GEO): Maximum Additional Payload Mass, Payload Power, and Maneuver ΔV That Can Be Attained With Innovative Technologies Launched on Delta II

No.	Technology	P/L Mass (kg)	% Fairing	P/L Power (kW)	% Fairing	Maneuver Δv (m/s)	% Fairing
Delta II Baseline Values							
1-1	Baseline Chemical	172	39	1.0	39	26	39
Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV							
1-2	Chemical	–5	39	0.0	39	–16	39
1-3	Advanced Cryo	+528	71	+2.8	71	+2845	74
1-4		+519	71	+2.8	71	+1148	74
1-5	Nuclear Bimodal	[a]	[a]	[a]	[a]	[a]	[a]
1-6		[a]	[a]	[a]	[a]	[a]	[a]
1-7		[a]	[a]	[a]	[a]	[a]	[a]
1-8	Solar Bimodal	+446	206	+3.7	206	+1329	206
1-9		+432	206	+3.7	206	+528	206
1-10		+443	206	+3.7	206	+985	206
1-11	Solar Thermal	+797	210	+4.3	210	+3892	210
1-12		+786	210	+4.3	210	+1577	210
1-13	Nuclear NH_3 Arcjet	[a]	[a]	[a]	[a]	[a]	[a]
1-14		[a]	[a]	[a]	[a]	[a]	[a]
1-15	Nuclear H_2 Arcjet	[a]	[a]	[a]	[a]	[a]	[a]
1-16		[a]	[a]	[a]	[a]	[a]	[a]
1-17	Nuclear Xe SPT	[a]	[a]	[a]	[a]	[a]	[a]
1-18		[a]	[a]	[a]	[a]	[a]	[a]
1-19	Nuclear Xe Ion	–47	184	–0.6	184	[a]	[a]
1-20		–81	184	–1.0	184	[a]	[a]
1-21	Solar NH_3 Arcjet	+262	84	+1.7	84	+1257	84
1-22		+249	84	+1.7	84	+367	84
1-23	Solar H_2 Arcjet	+480	187	+3.3	187	+1944	187
1-24		+469	187	+3.2	187	+780	187
1-25	Solar Xe SPT	+785	106	+5.5	106	+5528	103
1-26		+760	106	+5.3	106	+807	106
1-27	Solar Xe Ion	+1153	129	+8.4	129	+13486	126
1-28		+1120	129	+8.1	129	+989	129

[a]Launch vehicle capabilities exceeded: technology cannot support requirements. For *Max P/L Mass*, the payload power, moves, and MMD must equal that of the baseline; for *Max P/L Power* (or *Moves*), the payload mass, MMD and moves (or payload power) must equal that of the baseline (see discussion in MOP methodology).

Additional mass and power results for electric move from Table 5-3 through Table 5-5 are plotted in Figure 5-9 and Figure 5-10, respectively. The lift technologies are listed across the bottom of each figure and the additional payload mass (or electrical power) is given on the vertical axis. Each row of bars represents one launch vehicle as indicated: Titan in the rear (light colored bars), Atlas in the middle (dark bars), and Delta in front. Some of the additional masses are negative. They are indicated with zero-height bars. Missing bars indicate that the technology cannot place the nominal satellite on-orbit, even with a reduced payload mass (or electrical power).

Five lift technologies produce varying, but significant benefits with all three launch vehicles:

- Solar thermal
- Solar electric Xe SPT
- Solar electric Xe Ion
- Solar bimodal (to a lesser extent)
- Solar H_2 arcjet (also to a lesser extent)

Nuclear bimodal and nuclear electric Xe ion do well with Titan, moderately well with Atlas, and (except for solar electric) poorly with Delta. The poor performance of nuclear technologies with Delta II is caused by high reactor mass and Delta's low throw weight. Advanced cryogenic is also a consistent performer across launch vehicles, but typically at a lower level than the bulleted solar technologies.

Figures 5-11 and 5-12 show the tradeoff between additional payload mass and payload electrical power for Titan IV and Atlas IIAS, respectively. The figures assume zero additional ΔV. The baseline capabilities from Table 5-2 are indicated with a circled ***X***. A comparison of the two figures substantiates the earlier observation that nuclear technologies perform better relative to solar technologies on the larger Titan launch vehicle. The figures also shows that no technology on the Atlas IIAS can match the Titan baseline.

5.4.3.2 ORM 2A (MEO-GPS): ADDITIONAL CAPABILITIES PROVIDED BY INNOVATIVE TECHNOLOGIES

Table 5-6 presents the additional payload mass, payload power, and maneuver ΔV results for ORM 2a (MEO-GPS) for Atlas IIAS and Delta II. Currently Titan IV is not used to launch to GPS orbit. Technology combinations are again taken from Table 4-8. Figure 5-13 shows additional payload mass and Figure 5-14 shows additional payload electrical power. Again, additional ΔV is not plotted. With the exceptions of nuclear bimodal on the Atlas and advanced cryogenic on Delta, the standout performers are the solar technologies. This is further reinforced by Figure 5-15 and Figure 5-16 which trade payload mass and additional payload electrical power for zero additional ΔV.

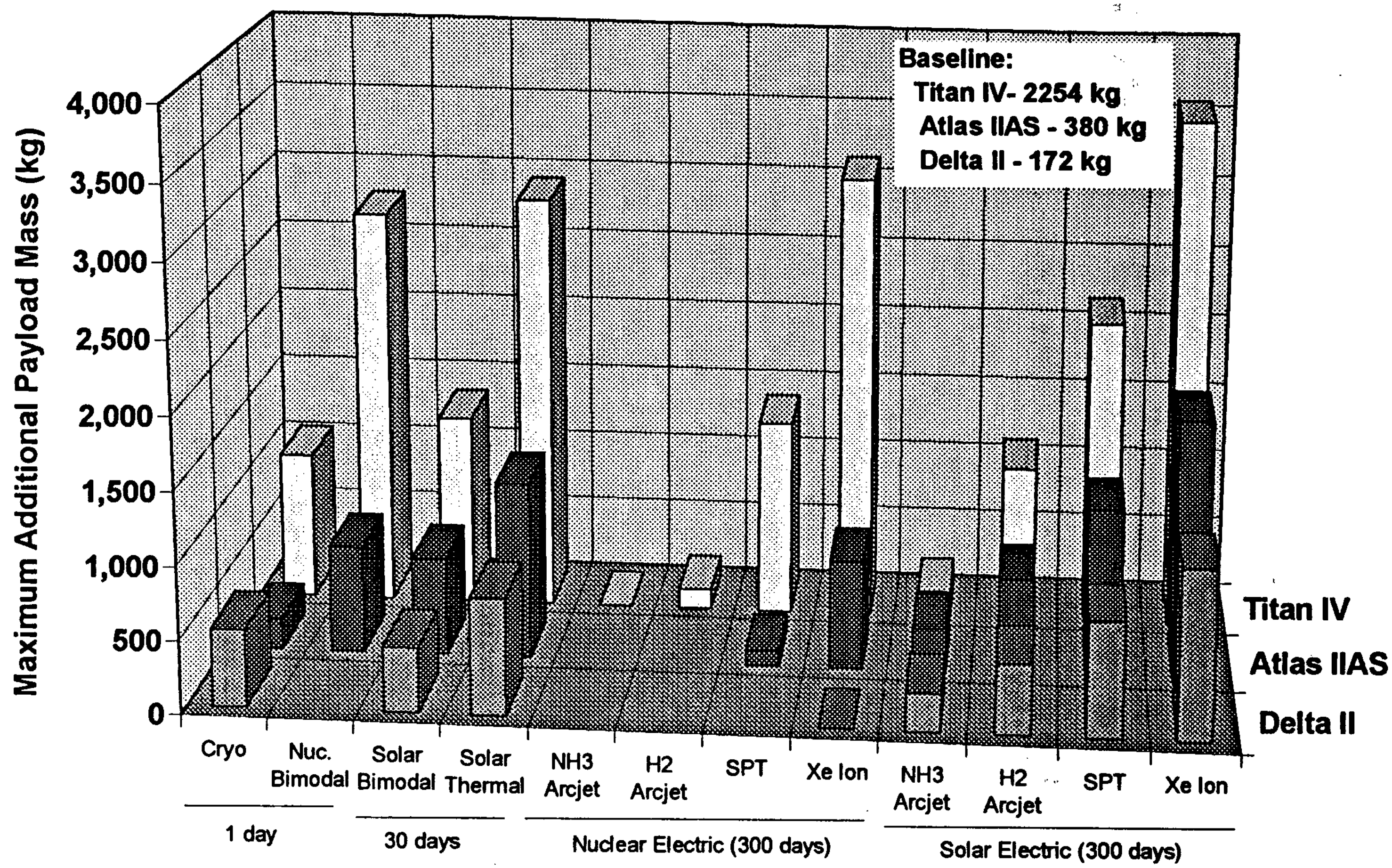


Figure 5-9. ORM 1 (GEO): maximum additional on-orbit payload mass that can be attained using innovative technologies launched on Atlas IIAS and Delta II.

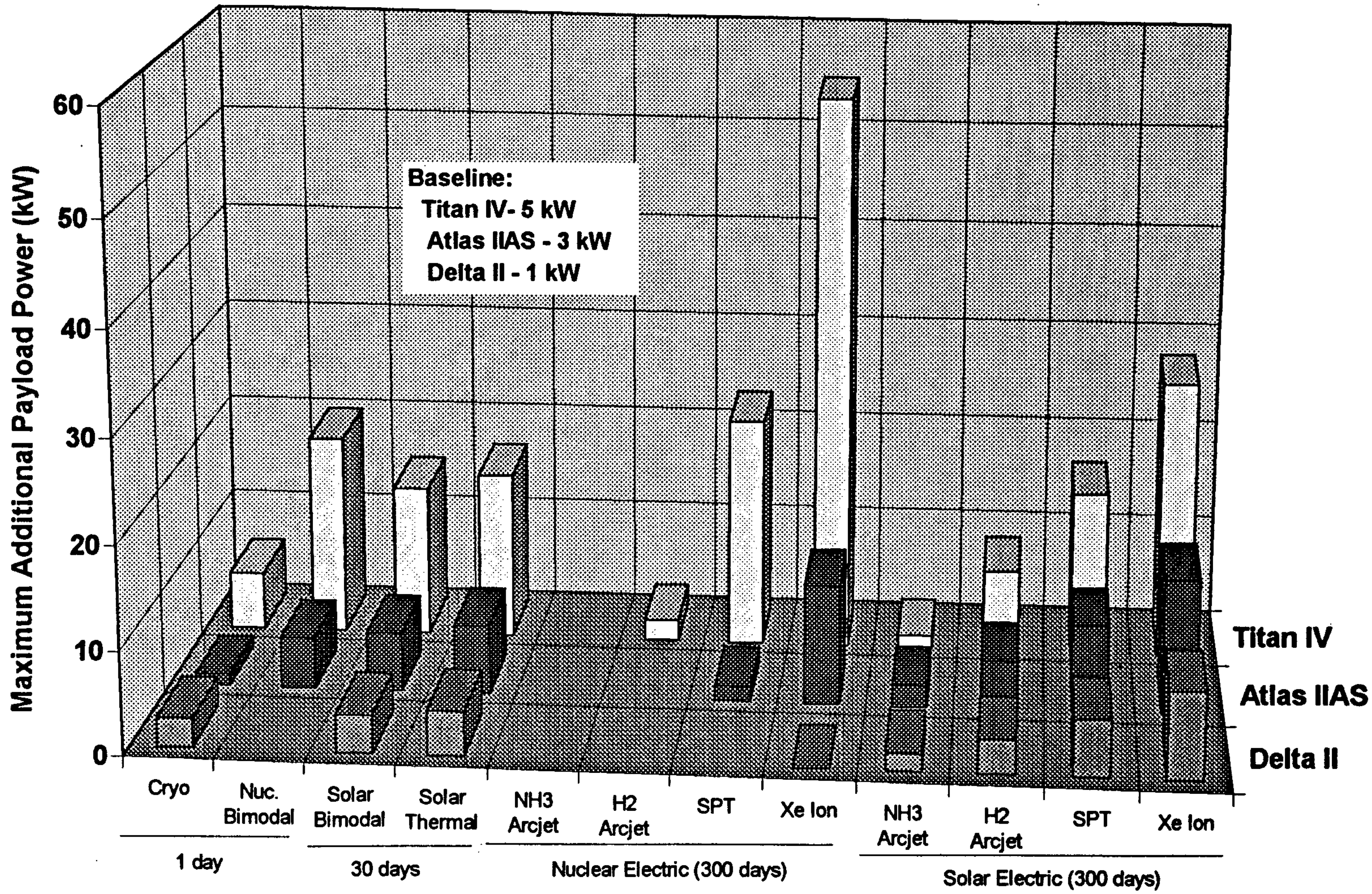


Figure 5-10. ORM 1 (GEO): maximum additional on-orbit payload electric power that can be attained using innovative technologies launched on Titan IV, Atlas IIAS, and Delta II.

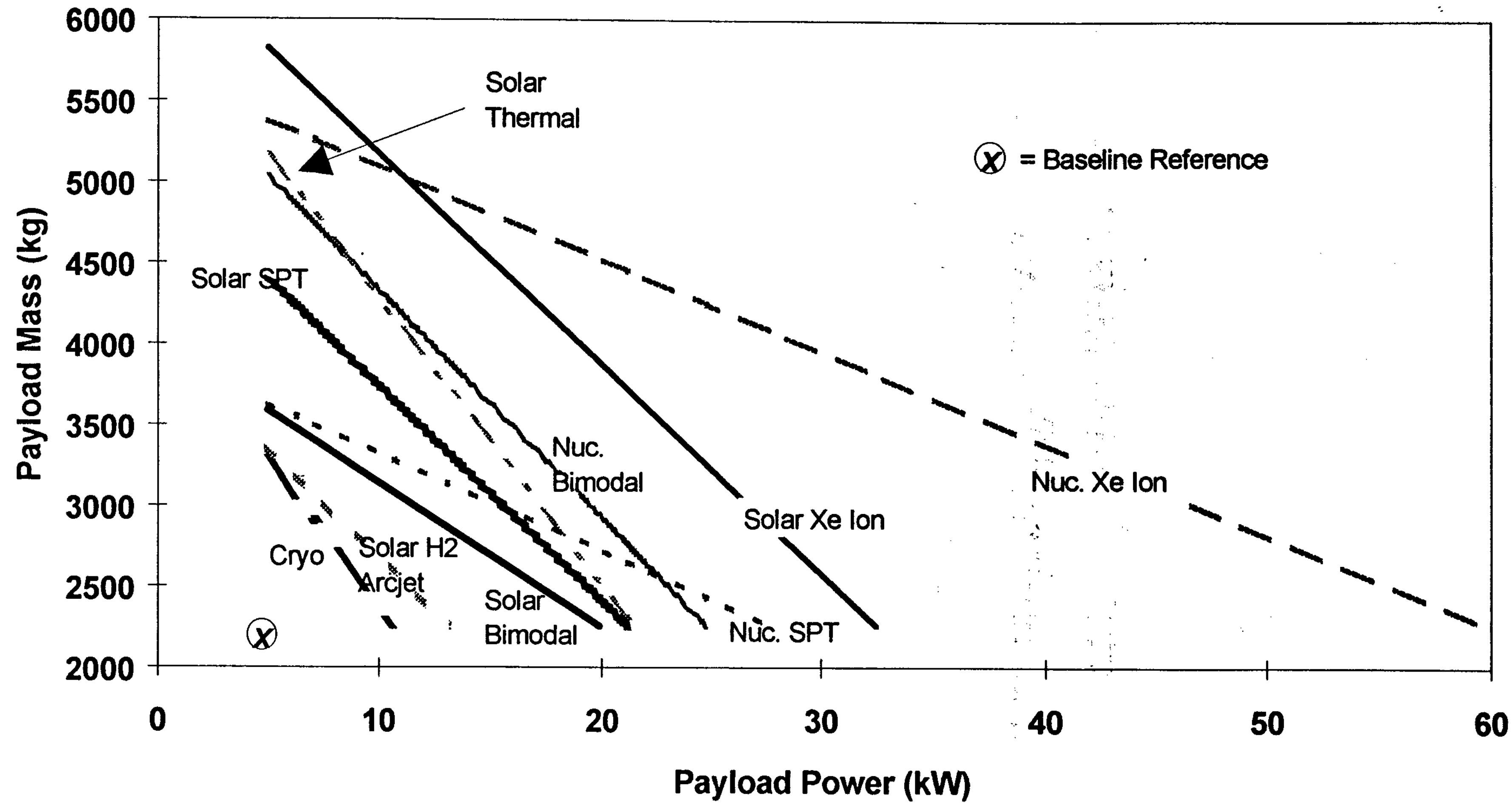


Figure 5-11. ORM 1 (GEO): trade-off of payload mass and payload electrical power for fixed maneuver ΔV using innovative technologies launched on Titan IV.

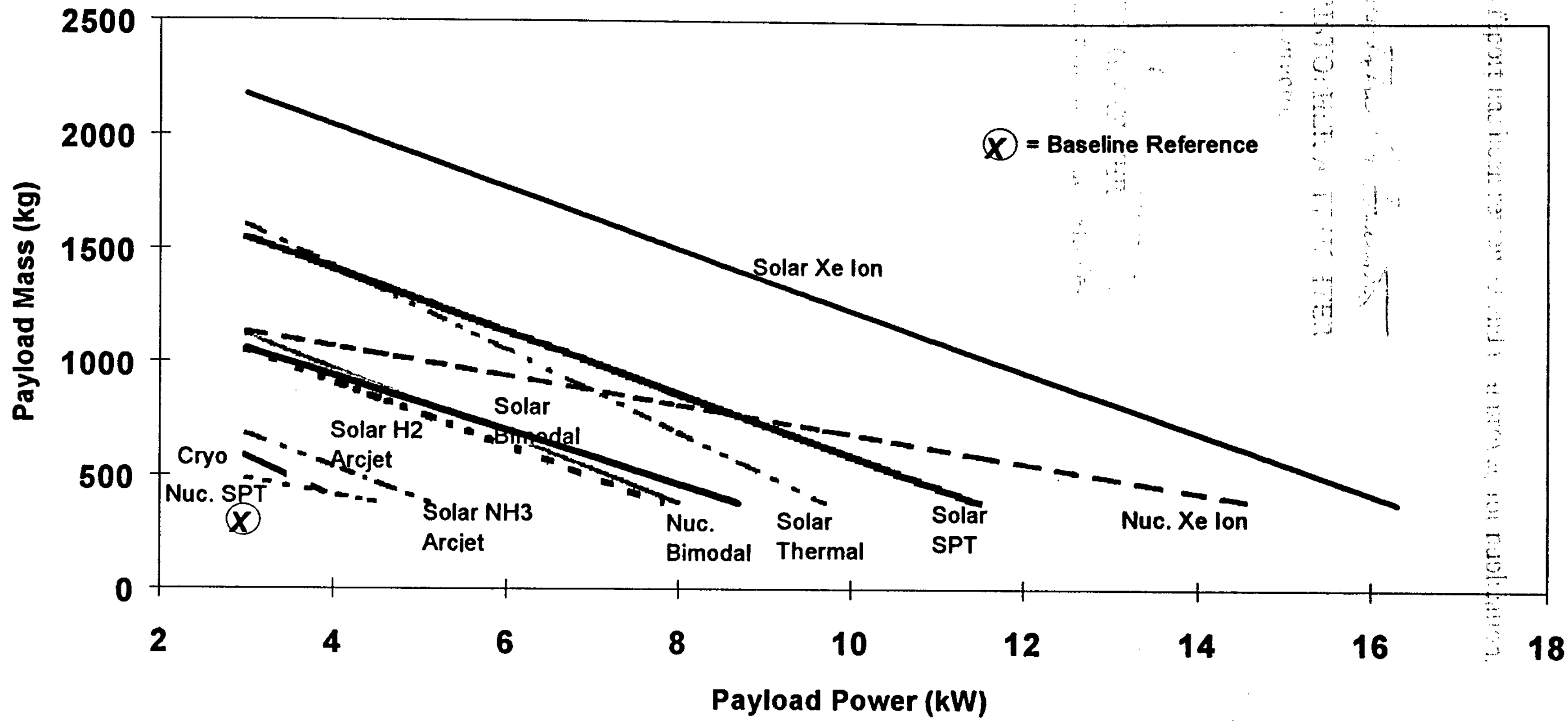


Figure 5-12. ORM 1 (GEO): trade-off of payload mass and payload electrical power for fixed maneuver ΔV using innovative technologies launched on Atlas IIAS.

Table 5-6. ORM 2a (GPS): Maximum Additional Payload Mass, Payload Power, and Maneuver ΔV That Can Be Attained With Innovative Technologies Launched on Atlas IIAS and Delta II

No.	Technology	P/L Mass (kg)	% Fairing	P/L Power (kW)	% Fairing	Maneuver Δv (m/s)	% Fairing
ATLAS IIAS: Baseline Values							
2-1	Baseline Chemical	1262	44	1.0	44	81	44
Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV							
2-2	Advanced Cryo	+350	53	+1.7	53	+467	53
2-3	Nuclear Bimodal	+901	165	6.0	165	+514	165
2-4	Solar Bimodal	+980	168	+7.0	171	+591	168
2-5	Solar Thermal	+1395	159	+7.2	159	+1146	159
2-6	Nuclear NH_3 Arcjet	–436	96	a	a	a	a
2-7	Nuclear H_2 Arcjet	–100	150	a	a	–51	150
2-8	Nuclear Xe SPT	+62	115	+0.8	115	+29	115
2-9	Nuclear Xe Ion	+351	138	+5.0	138	+152	138
2-10	Solar NH_3 Arcjet	+702	78	+4.8	78	+476	78
2-11	Solar H_2 Arcjet	+1059	129	+7.4	129	+737	129
2-12	Solar Xe SPT	+1263	91	+8.9	91	+705	91
2-13	Solar Xe Ion	+1534	101	+11.1	101	+760	101
DELTA II: Baseline Values							
2-1	Baseline	387	47	1.0	47	81	47
Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV							
2-2	Advanced Cryo	+772	90	+4.1	90	+1613	90
2-3	Nuclear Bimodal	a	a	a	a	a	a
2-4	Solar Bimodal	+707	210	+4.3	197	+773	210
2-5	Solar Thermal	+891	197	+4.7	197	+1407	197
2-6	Nuclear NH_3 Arcjet	–348	145	a	a	a	a
2-7	Nuclear H_2 Arcjet	–210	216	a	a	a	a
2-8	Nuclear Xe SPT	–158	165	a	a	a	a
2-9	Nuclear Xe Ion	–59	184	–0.8	184	–41	184
2-10	Solar NH_3 Arcjet	+607	103	+4.2	103	+713	103
2-11	Solar H_2 Arcjet	+793	171	+5.6	171	+994	171
2-12	Solar Xe SPT	+956	119	+6.8	119	+920	119
2-13	Solar Xe Ion	+1134	135	+8.2	135	+966	135

[a]Launch vehicle capabilities exceeded: technology cannot support requirements. For *Max P/L Mass*, the payload power, moves, and MMD must equal that of the baseline; for *Max P/L Power* (or *Moves*), the payload mass, MMD and moves (or payload power) must equal that of the baseline (see discussion in MOP methodology).

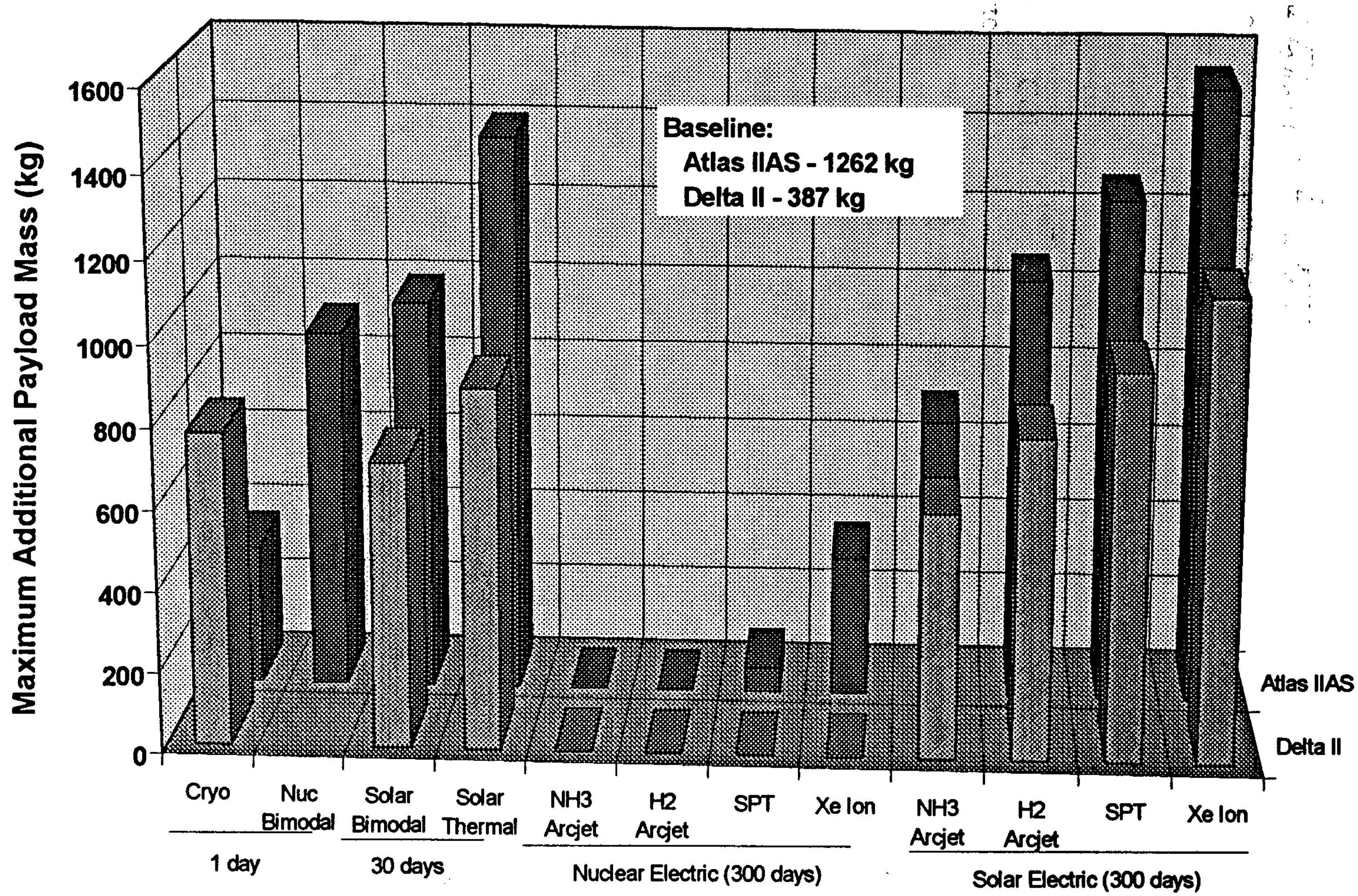


Figure 5-13. ORM 2a (GPS): maximum additional on-orbit payload mass that can be attained using innovative technologies launched on Atlas IIAS and Delta II.

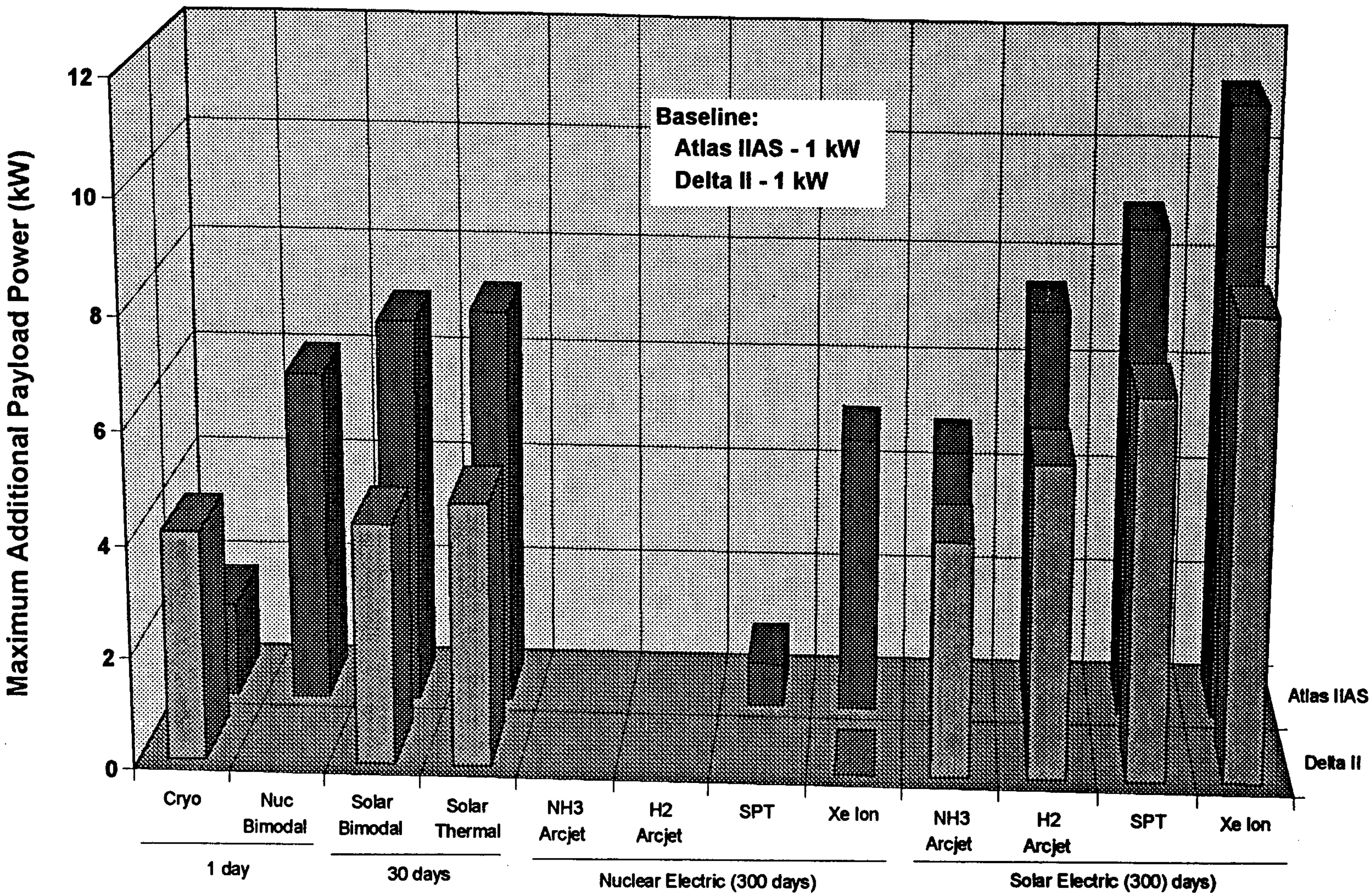


Figure 5-14. ORM 2a (GPS): maximum additional on-orbit payload electric power that can be attained using innovative technologies launched on Atlas IIAS and Delta II.

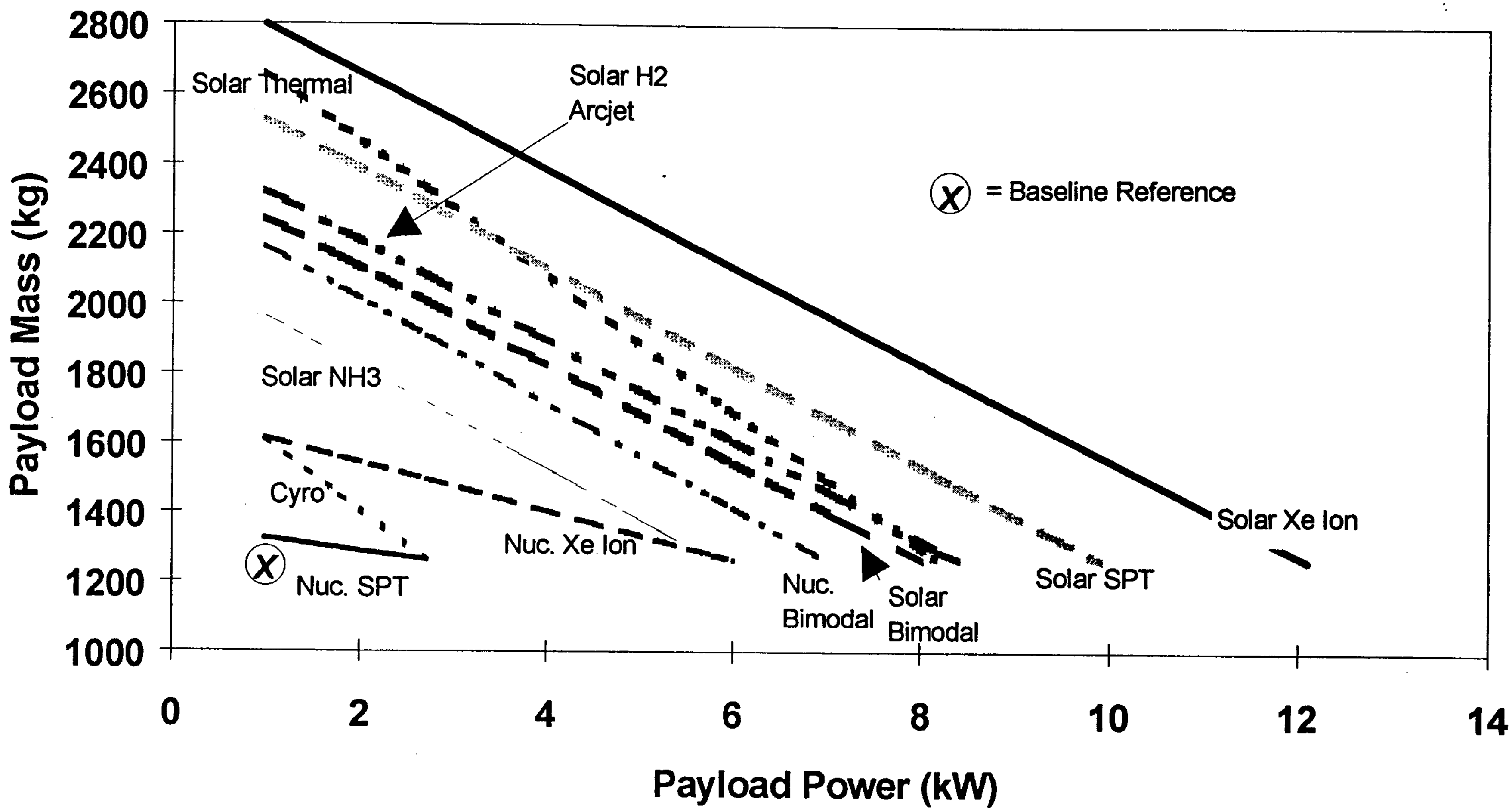


Figure 5-15. ORM 2a (GPS): payload mass and payload electrical power for fixed maneuver ΔV using innovative technologies launched on Atlas IIAS.

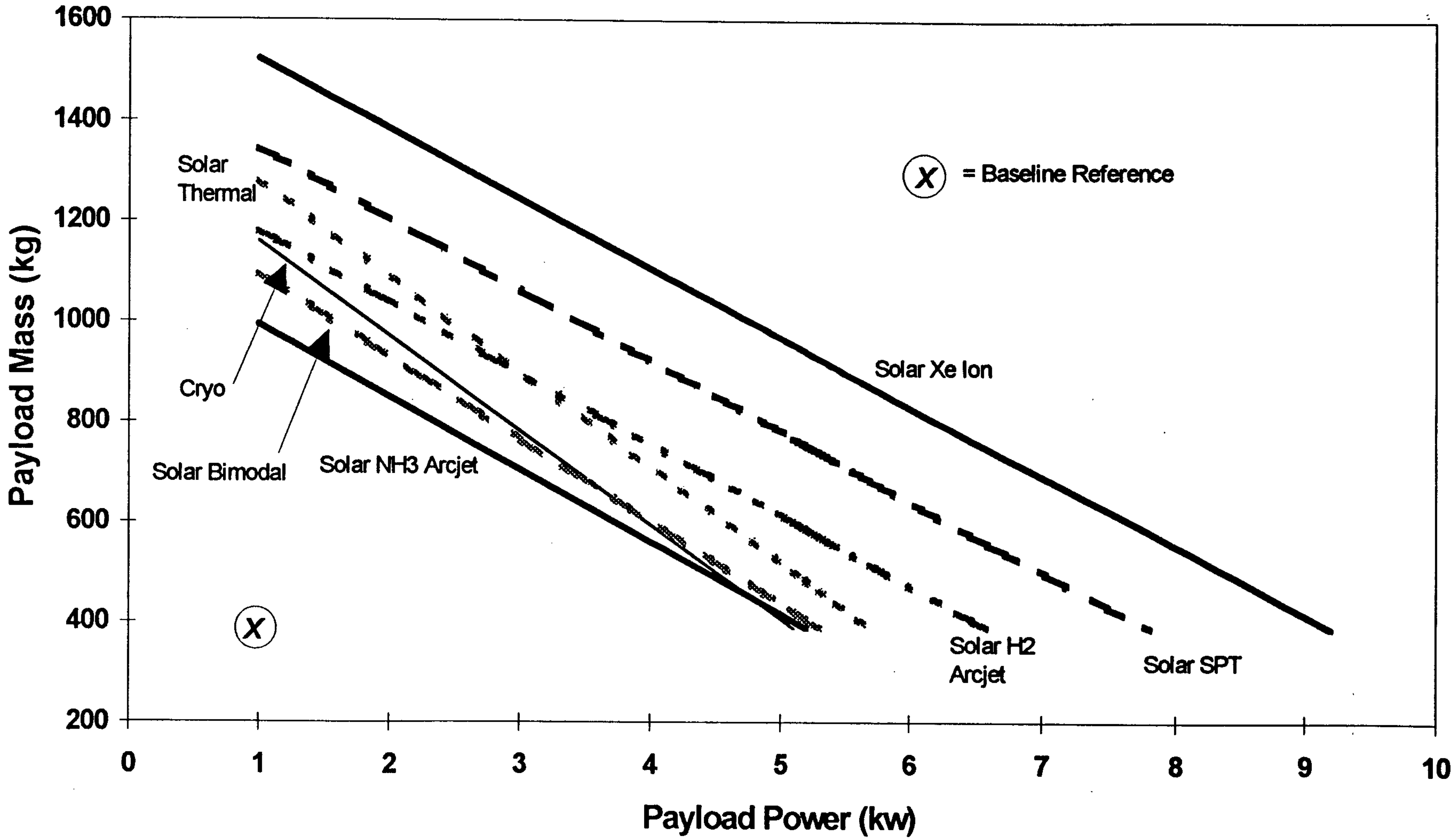


Figure 5-16. ORM 2a (GPS): trade-off between limiting payload mass and payload electrical power for fixed ΔV using innovative technologies launched on Delta II.

5.4.3.3 ORM 3A (LEO-POLAR): ADDITIONAL CAPABILITIES PROVIDED BY INNOVATIVE TECHNOLOGIES

Table 5-7 show the results of the additional capability for ORM 3a (LEO-polar) using the Atlas IIAS, Delta II, and Lockheed Launch Vehicle 3 (LLV3). We have included LLV3 because it has a known baseline technology capability and is considered later in this chapter as a step-down option for Delta. The innovative technologies have little opportunity to take advantage of their higher I_{sp} values because of the small ΔV requirements for ORM 3a to raise the orbit a few hundred kilometers. Thus improvements in on-orbit payload mass, payload electrical power, and ΔV are minimal except for advanced cryogenic with Delta. The improvement for advanced cryogenic with Delta comes from substituting the cryogenic upper stage for the second-stage and upper-stage PAM. Because the additional capabilities for ORM 3a are generally minimal and the technologies few, we have not plotted the results.

5.4.3.4 ORM 4 (HEO): ADDITIONAL CAPABILITIES PROVIDED BY INNOVATIVE TECHNOLOGIES

The semimajor axes, hence energies, of ORM 4 (HEO) and ORM 2a (MEO-GPS) are nearly identical. Thus it is to be expected that the same technologies will show comparable results for the two ORMs. The results of the additional capability for HEO are presented in Table 5-8. If we compare Table 5-6 and Table 5-8, we see the same technologies do have essentially comparable results. Because of the similar increases in performance and the limited number of technologies for ORM 4 (HEO), we have not plotted the results from Table 5-8. There are fewer technologies for ORM 4 (HEO) because the electric technologies are not well suited for transfers to HEO with its large orbital eccentricity.

5.4.3.5 SUMMARY OF ADDITIONAL CAPABILITIES PROVIDED BY INNOVATIVE TECHNOLOGIES

Figures 5-17 and 5-18 summarize by ORM and technology the additional payload mass and electrical power that can be placed in mission orbit using the innovative technologies. The ORMs are organized by launch vehicle at the top of each figure. The baseline capability from Table 5-2 is shown for reference below each ORM. The technologies are identified on the left of each figure. All the innovative lift technologies are represented, but only the best performing of the subtechnologies are shown. The summary results consist of the multipliers of the baseline values corresponding to each ORM-technology pair. For example, for a 5 kW power payload (see Table 5.2), Titan IV can lift a baseline payload mass of 2254 kg to GEO. In contrast, if the solar thermal innovative technology were used, Titan IV could lift approximately 2.3 x 2254 kg (as indicated by the 2.3 appearing in the solar thermal/Titan IV/GEO box).

All multipliers greater than 1.0 are given numerically, with multipliers greater than 2.0 appearing in the more heavily shaded boxes. Combinations performing more poorly than the baseline are indicated by blank, heavily shaded boxes. The lighter shaded blank boxes indicate the technologies are unsuitable for the corresponding ORMs.

Table 5-7. ORM 3a (LEO-Polar): Maximum Additional Payload Mass, Payload Power, and Maneuver ΔV That Can Be Attained With Innovative Technologies Launched on Atlas IIAS, Delta II, and LLV3

No.	Technology	P/L Mass (kg)	% Fairing	P/L Power (kW)	% Fairing	Maneuver Δv (m/s)	% Fairing
	ATLAS IIAS: Baseline Values						
3-1	Baseline Chemical	3843	80	2.0	117	120	117
	Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV						
3-2	Direct	−379	107	−1.3	107	[a]	[a]
3-3	Cryo	+341	126	+1.2	126	+143	126
3-5	Solar H_2 Arcjet	+36	118	−0.6	118	+16	118
3-6	Solar NH_3 Arcjet	+86	113	−0.4	113	+37	113
3-7	Solar N_2H_4 Arcjet	+41	112	−0.6	112	+18	112
3-8	Solar SPT	+67	115	−0.4	115	+28	115
3-9	Solar Xe Ion	+48	115	−0.5	115	+20	150
	DELTA II: Baseline Values						
3-1	Baseline Chemical	1890	130	1.2	130	20	130
	Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV						
3-2	Direct	−38	126	−0.1	126	[a]	[a]
3-3	Cryo	+1978	232	+7.6	232	+1169	239
3-5	Solar H_2 Arcjet	+32	139	+0.2	135	+25	139
3-6	Solar NH_3 Arcjet	+98	132	+0.4	132	+75	132
3-7	Solar N_2H_4 Arcjet	+52	129	+0.2	129	+40	129
3-8	Solar SPT	+102	135	+0.5	132	+77	135
3-9	Solar Xe Ion	+76	135	+0.4	135	+56	135
	LLV3: Baseline Values						
3-1	Baseline Chemical	1304	56	1.2	56	20	56
	Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV						
3-2	Direct	−740	51	−0.5	51	[a]	[a]
3-3	Cryo	[b]	[b]	[b]	[b]	[b]	[b]
3-5	Solar H_2 Arcjet	−561	60	+0.1	60	+26	60
3-6	Solar NH_3 Arcjet	−489	56	+0.4	56	+96	56
3-7	Solar N_2H_4 Arcjet	−525	56	+0.3	56	+61	56
3-8	Solar SPT	−481	58	0.5	58	+103	58
3-9	Solar Xe Ion	−499	58	+0.5	58	+84	58

[a]Launch vehicle capabilities exceeded: technology cannot support requirements. For *Max P/L Mass*, the payload power, moves, and MMD must equal that of the baseline; for *Max P/L Power* (or *Moves*), the payload mass, MMD and moves (or payload power) must equal that of the baseline (see discussion in MOP methodology).
[b]OECS has no LLV3 lift propulsion design for comparison.

Table 5-8. ORM 4 (HEO): Maximum Additional Payload Mass, Payload Power, and Maneuver ΔV That Can Be Attained With Innovative Technologies Launched on Titan IV, Atlas IIAS, and Delta II

No.	Technology	P/L Mass (kg)	% Fairing	P/L Power (kW)	% Fairing	Maneuver ΔV (m/s)	% Fairing
TITAN IV: Baseline Values							
4-1	Baseline Direct[a]	1665	14	2.8	14	190	14
Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV							
4-2	Direct	+65	14	+0.3	14	+130	14
4-3	Cryo	[b]	[b]	[b]	[b]	[b]	[b]
4-5	Nuclear Bimodal	+4556	83	+35.3	83	+3567	83
4-7	Solar Bimodal	+3828	76	+27.2	70	+2867	76
4-9	Solar Thermal	+4726	72	+28.7	72	+4695	72
ATLAS IIAS: Baseline Values							
4-1	Direct	852	49	2.8	49	190	49
Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV							
4-2	Direct	+77	50	+0.4	50	+221	50
4-3	Cryo	+367	57	+2.0	57	+964	57
4-5	Nuclear Bimodal	+787	154	+5.7	154	+1239	154
4-7	Solar Bimodal	+998	157	+6.5	159	+1682	157
4-9	Solar Thermal	+1194	149	+6.9	149	+2527	149
DELTA II: Baseline Values							
4-1	Direct	365	48	1.0	48	95	48
Maximum Additional P/L Mass, P/L Power, and Maneuver ΔV							
4-2	Direct	–5	48	0.0	48	–31	48
4-3	Cryo	+663	90	+3.8	90	+2777	90
4-5	Nuclear Bimodal	[c]	[c]	[c]	[c]	[c]	[c]
4-7	Solar Bimodal	+646	197	+4.2	184	+1892	197
4-9	Solar Thermal	+794	184	+4.6	184	+3134	184

[a]Titan IV does not go directly into HEO, but into an orbit with a slightly lower perigee which is raised using an integral bipropellant system. Atlas and Delta go directly into HEO.

[b]A cryo stage is not used with the Titan IV (see note a).

[c]Launch vehicle capabilities exceeded: technology cannot support requirements. For *Max P/L Mass*, the payload power, moves, and MMD must equal that of the baseline; for *Max P/L Power* (or *Moves*), the payload mass, MMD and moves (or payload power) must equal that of the baseline (see discussion in MOP methodology).

	Titan IV		Atlas IIAS				Delta II				LLV3
	GEO	HEO	GEO	MEO	LEO	HEO	GEO	MEO	LEO	HEO	LEO
Baseline (kg)	2254	1665	380	1262	3843	852	172	387	1890	365	1304
Adv. Cryo	1.5		1.6	1.3	1.1	1.4	[illegible]	3.0	2.1	[illegible]	a
Nuc. Bimodal	2.2	3.7	2.9	1.7		1.9					
Solar Bimodal	1.6	3.3	2.8	1.8		2.2	3.6	2.8		[illegible]	
Solar Thermal	2.3	3.8	4.2	2.1		2.4	3.6	3.9		[illegible]	
Nuclear Ion	2.4		3.0	1.3							
Solar H_2 Arcjet	1.1		2.8	1.6	1.0		3.8	3.0	1.0		1.0
Solar SPT	2.0		4.1	2.2	1.0		5.6	3.5	1.1		1.1
Solar Ion	2.6		5.7	2.2	1.0		7.7	3.9	1.0		1.1

>2.0 x baseline   # 1.0–2.0 x baseline   < Baseline   Technology Not Suitable

[a]No LLV3 cryo lift design for comparison

Figure 5-17. Summary of the multiple of the baseline payload mass that can be placed in mission orbit using innovative technologies.

	Titan IV		Atlas IIAS				Delta II				LLV3
	GEO	HEO	GEO	MEO	LEO	HEO	GEO	MEO	LEO	HEO	LEO
Baseline (kW)	5.0	2.75	3.0	1.0	2.0	2.75	1.0	1.0	1.25	1.0	1.25
Adv. Cryo	2.1		1.4	2.7	1.6	1.7	3.8	[illegible]	[illegible]	[illegible]	a
Nuc. Bimodal	5.0	13.8	2.7	7.0		3.1					
Solar Bimodal	4.0	10.9	2.9	8.0		3.4	4.7	5.8		[illegible]	
Solar Thermal	4.3	11.4	3.2	8.2		3.5	5.3	[illegible]		[illegible]	
Nuclear Ion	12.0		4.9	6.0							
Solar H_2 Arcjet	2.6		2.6	8.4			4.8	6.6	1.2		1.1
Solar SPT	4.2		3.8	9.9			6.5	7.8	1.4		1.4
Solar Ion	6.5		5.4	12.1			9.4	9.2	1.3		1.4

\# >2.0 x baseline \# 1.0–2.0 x baseline < Baseline Technology Not Suitable

[a]No LLV3 cryo lift design for comparison

Figure 5-18. Summary of the multiple of the baseline payload electric power that can be placed in mission orbit using innovative technologies.

In a pattern evident throughout the effectiveness summaries, the solar technologies perform well across all ORMs and launch vehicles, while nuclear bimodal and nuclear electric perform competitively on Titan and Atlas. Solar ion is generally the best performer. As mentioned earlier, advanced cryogenic performs very well on the Delta because it is replacing the second stage and the relatively small PAM chemical upper stage.

5.4.3.6 MOP 2-3: IMPACTS OF PROPULSION AND POWER SUBSYSTEMS ON THE SATELLITE

Figure 3-1 in Chapter 3 identified which OECS technologies remain with the satellite after lift and which separate. The baseline chemical and advanced cryogenic upper stages separate. The same is true for solar thermal stages, and at least some of the arrays in the solar electric stage are likely to be discarded. However, the nuclear and solar bimodal and nuclear electric stages are integral with the satellite and do not separate.

In light of these differences, the innovative technologies are sure to have some impact, perhaps substantial, on both satellite and payload designs. While it was not within the scope of the OECS to examine these impacts, Table 5-9 summarizes some possibilities. An X indicates a potential impact, but in no way implies that a serious problem exists. Only detailed engineering and testing can provide reliable assessments of impact.

Table 5-9. Potential Impacts of Innovative Technologies on Selected Areas

	Innovative Technology					
Impact	Advanced Cryogenic	Nuclear Bimodal	Solar Bimodal	Solar Thermal	Nuclear Electric	Solar Electric
Payload Prelaunch Handling		X	X		X	
Payload Placement on Satellite		X	X		X	
Satellite Radiation Environment		X			X	
Satellite Thermal Environment		X	X		X	
Payload Design		X	X		X	X
Satellite On-Orbit Dynamics		X	X		X	

5.5 MOE 3: ALTERNATIVE LAUNCH APPROACHES

5.5.1 Background and MOPs

MOE 3 examines the potential of the innovative technologies to off-load payloads from the baseline-technology launch vehicles to smaller launch vehicles. This is known as *step-down*. Step-down is desirable because launch vehicle cost decreases rapidly with decreasing capability (see Table 6.2). We examine the following specific step-downs: Titan to Atlas, Titan to Delta, Atlas to Delta, and Delta to LLV3. Additionally, we make point estimates of the capability needed by hypothetical new launch vehicles using innovative upper stages to replace the existing baseline capability.

We have defined three MOPs to support MOE 3:

- MOP 3-1: Potential for payload step-down
- MOP 3-2: Sizing of hypothetical future launch vehicles
- MOP 3-3: Impact of step-down on launch vehicles

The first MOP determines what fraction of a given representative payload can be stepped down. The second indicates what capabilities might be reasonable in new launch vehicles using innovative technologies. The third determines the approximate fairing volumes associated with each step-down.

5.5.2 Methodology and Results

MOP 3-1 and MOP 3-2 are discussed separately below. Our data on MOP 3-3 are integrated into both discussions. The discussion of how we estimated the required fairing volumes was presented earlier in this chapter (Section 5.3.3.1).

5.5.2.1 MOP 3-1: STEP-DOWN POTENTIAL OF INNOVATIVE TECHNOLOGIES

We define *step-down potential* as "the percentage of the payload mass defined in Table 5-2 that can be stepped down to a smaller launch vehicle." For example, the Titan-Centaur can place 2254 kg of payload mass into ORM 1 (GEO). If an innovative upper stage technology atop Atlas could deliver 2000 kg of payload mass with identical payload electrical power and maneuver ΔV, the corresponding step-down potential would be 89%. We assume that payload electrical power, MMD, and maneuver ΔV from Table 5-2 are unchanged when determining the percentage payload mass that can be stepped down. This definition is one of several step-down measures that were considered for MOP 3-1. It was selected because it is straightforward and easily understood.

Step-down potential is calculated with the OCEM model in bottom-up mode. Percentages below 100% indicate complete payload step-down is not possible. Percentages greater than 100% indicate that more than complete step-down is possible. Tables 5-10—5-13 present the step-down results for ORMs 1, 2a, 3a, and 4.

Table 5-10. ORM 1 (GEO): Percent of Baseline Payload That Can Be Stepped Down Using Innovative Technologies

		Titan IV Baseline				Atlas IIAS Baseline		Delta II Baseline	
		Payload 2254 kg Payload Pwr 5.0 kW EOL ΔV 51.6 m/s MMD 10 yr				Payload 380 kg Payload Pwr 3.0 kW EOL ΔV 51.6 m/s MMD 10 yr		Payload 172 kg Payload Pwr 1.0 kW EOL ΔV 25.8 m/s MMD 7 yr	
		Titan IV to Atlas IIAS		Titan IV to Delta II		Atlas IIAS to Delta II		Delta II to LLV3	
No.	Technology	% Step-down[a]	% Fairing[b]	% Step-down[a]	% Fairing[b]	% Step-down[a]	% Fairing[b]	% Step-down[a]	% Fairing[b]
1-3	Advanced Cryo	9	44	0	0	62	73	[c]	[c]
1-5	Nuclear Bimodal	36	174	0	0	0	0	[d]	[d]
1-8	Solar Bimodal	39	180	1	208	84	207	150	95
1-11	Solar Thermal	54	168	5	210	130	210	364	91
1-13	Nuclear NH_3 Arcjet	0	0	0	0	0	0	0	0
1-15	Nuclear H_2 Arcjet	0	0	0	0	0	0	0	0
1-17	Nuclear Xe SPT	15	106	0	0	0	0	0	0
1-19	Nuclear Xe Ion	44	137	0	0	0	0	0	0
1-21	Solar NH_3 Arcjet	18	64	0	0	16	84	15	26
1-23	Solar H_2 Arcjet	34	140	0	0	70	186	72	59
1-25	Solar Xe SPT	56	81	13	106	156	106	174	34
1-27	Solar Xe Ion	84	98	31	130	257	130	302	41

[a]Percent of payload mass that can be stepped down while holding electrical power, maneuver ΔV, and MMD unchanged.
[b]Percent of target vehicle's largest fairing volume.
[c]OECS has no LLV3 lift propulsion designs for comparison.
[d]Nuclear bimodal does not fit on a Delta.

Table 5-11. ORM 2a (GPS-MEO): Percent of Baseline Payload That Can Be Stepped Down Using Innovative Technologies

		Atlas IIAS Baseline		Delta II Baseline	
		Payload 1262 kg Payload Power 1.0 kW EOL ΔV 80.6 m/s MMD 10 yr		Payload 387 kg Payload Power 1.0 kW EOL ΔV 80.6 m/s MMD 10 yr	
		Atlas IIAS to Delta II		Delta II to LLV3	
No.	Technology	% Step-down[a]	% Fairing[b]	% Step-down[a]	% Fairing[b]
2-2	Advanced Cryo	91	90	[c]	[c]
2-3	Nuclear Bimodal	51	234	[d]	[d]
2-4	Solar Bimodal	87	210	159	96
2-5	Solar Thermal	100	197	212	83
2-6	Nuclear NH_3 Arcjet	3	146	0	0
2-7	Nuclear H_2 Arcjet	14	216	0	0
2-8	Nuclear Xe SPT	18	165	0	0
2-9	Nuclear Xe Ion	26	185	0	0
2-10	Solar NH_3 Arcjet	78	102	109	34
2-11	Solar H_2 Arcjet	93	172	129	58
2-12	Solar Xe SPT	106	120	159	40
2-13	Solar Xe Ion	120	134	189	45

[a]Percent of payload mass that can be stepped down while holding electrical power, maneuver ΔV, and MMD unchanged.
[b]Percent of target vehicle's largest fairing volume.
[c]OECS has no LLV3 lift propulsion designs for comparison.
[d]Nuclear bimodal does not fit on a Delta.

The nature of the step-down is indicated at the top of each table along with a description of the nominal payload characteristics from Table 5-2. Only the best performing technology combination for each lift technology is represented. Estimated combined upper stage and satellite volumes are compared with the smaller launch vehicle's largest fairing volumes.

ORM 1 (GEO)

From Table 5-10 we find the following:

- While we cannot completely replace Titan IV with Atlas for any innovative technology (84% step-down for solar Xe ion), several technologies can step down more than 50% of the mass of a Titan IV payload.

Table 5-12. ORM 3a (LEO-Polar): Percent of Baseline Payload That Can Be Stepped Down Using Innovative Technologies

		Atlas IIAS Baseline		Delta II Baseline	
		Payload 3843 kg Payload Power 2.0 kW EOL ΔV 120 m/s MMD 7 yr		Payload 1890 kg Payload Power 1.3 kW EOL ΔV 20 m/s MMD 5.5 yr	
		Atlas IIAS to Delta II		Delta II to LLV3	
No.	Technology	% Step-down	% Fairing[a]	% Step-down	% Fairing[a]
3-2	Direct	39	127	61	51
3-3	Cryo	89	234	[b]	[b]
3-5	Solar NH_3 Arcjet	48	132	74	57
3-6	Solar H_2 Arcjet	46	137	70	59
3-7	Solar N_2H_4 Arcjet	47	130	72	56
3-8	Solar SPT	48	134	75	56
3-9	Solar Xe Ion	47	134	74	58

[a]Percent of target vehicle's fairing volume.
[b]OECS had no LLV3 lift propulsion design for comparison.

- A number of innovative technologies can provide complete step-down from Atlas IIAS to Delta II and Delta II to LLV3.
- The Delta fairing appears too small to accommodate Atlas IIAS to Delta step-down.
- The LLV3 has a large fairing and appears to accommodate the technologies required for step-down from Delta II.
- The most promising technologies are solar bimodal, solar thermal, and solar electric (although all have potential fairing volume problems).
- Nuclear technologies are not suited to step down to the smaller launch vehicles.

ORM 2a (GPS-MEO)

From ORM 2a (GPS-MEO) in Table 5-11, we see only Atlas to Delta and Delta to LLV3 step-downs are of interest. (Titan is not required for any of the OECS parametric payloads.) In both instances there are innovative technologies that appear capable of complete step-down, although fairing volumes are a potential issue in the Atlas to Delta case. The solar technologies are again the most promising.

ORM 3a (LEO-Polar)

The results of Table 5-12 show some step-down potential, but no innovative technology can provide complete step-down. The relatively low ΔV required to place the payload into a LEO polar orbit from parking orbit severely reduces the advantage of the higher I_{sp}'s of the innovative technologies.

Table 5-13. ORM 4 (HEO): Percent of Baseline Payload That Can Be Stepped Down Using Innovative Technologies

		Titan IV Baseline				Atlas IIAS Baseline		Delta II Baseline	
		Payload 1665 kg Payload Pwr 2.8 kW EOL ΔV 190 m/s MMD 7.5 yr		Payload 1665 kg Payload Pwr 2.8 kW EOL ΔV 190 m/s MMD 7.5 yr		Payload 852 kg Payload Pwr 2.8 kW EOL ΔV 190 m/s MMD 7.5 yr		Payload 365 kg Payload Pwr 1.0 kW EOL ΔV 95 m/s MMD 5 yr	
		Titan IV to Atlas IIAS		Titan IV to Delta II		Atlas IIAS to Delta II		Delta II to LLV3	
No.	Technology	% Step-down	% Fairing[a]	% Step-down	% Fairing[a]	% Step-down	% Fairing[a]	% Step-down	% Fairing[a]
4-2	Direct	56	49	0	48	0	48	6	15
4-3	Cryo	[b]	[b]	[b]	[b]	77	91	[c]	[c]
4-5	Nuclear Bimodal	98	155	17	222	33	222	[d]	[d]
4-7	Solar Bimodal	111	157	47	196	93	196	140	87
4-9	Solar Thermal	123	149	47	185	92	185	186	75

[a]Percent of target vehicle's largest fairing volume.
[b]Cryo is not used with Titan IV; step-down with this technology is not applicable.
[c]OECS had no LLV3 lift propulsion designs for comparison.
[d]Nuclear bimodal does not fit on Delta or LLV3.

ORM 4 (HEO)

Table 5-13 shows several of the innovative technologies can provide Titan IV with complete step-down for ORM 4 (HEO), unlike the situation for ORM 1 (GEO). This occurs because Titan does not employ Centaur in transferring to HEO, whereas Atlas does. This narrows the performance gap between the two launchers and enables the step-down. Solar bimodal and solar thermal realize almost 50% step-down to Delta from Titan and over 100% from Delta to LLV3.

5.5.2.2 SUMMARY OF STEP-DOWN POTENTIAL OF INNOVATIVE TECHNOLOGIES

Figure 5-19 summarizes the step-down potential shown in Table 5-10 through Table 5-13. The format of the figure is similar to that used to summarize additional payload mass and electrical power in Section 5.4. The step-down is identified at the top of the Figure 5-19 (e.g., Titan to Atlas). Titan to Delta is omitted due to lack of significant step-down potential. The relevant ORMs are indicated below the step-down. The abbreviations for the ORMs—GEO, MEO, LEO, and HEO—correspond to ORMs 1 (GEO), 2a (MEO-GPS), 3a (LEO-Polar), and 4 (HEO). Immediately below the ORM designations are the payload masses of the representative baseline technology, which are repeated from Table 5-2. These are the masses on which the results are based. The lift technologies are listed on the left. Only the top-performing nuclear electric technology and the three top-performing solar electric technologies are included in the summary.

A summary box is located at the intersection of each technology row and ORM column. Boxes containing numbers indicate the step-down potential as the ratio

$$\frac{\text{step - down payload mass}}{\text{baseline - technology payload mass}}$$

where the appropriate step-down payload mass from the aforementioned tables represents the best result for each technology. Ratios less than one indicate partial payload step-down; ratios greater than one indicate complete step-down with remaining vehicle lift margin or, alternatively, greater payload mass in the indicated ratio. Darkly shaded blank boxes indicate no step-down is possible; lighter-shaded blank boxes indicate the technology is not suitable for the indicated combination.

Looking across all possible step-downs and ORMs, the solar technologies appear to be the best performers. Nuclear bimodal is competitive for Titan to Atlas step-down only. The good performance of advanced cryogenic for Atlas IIAS to Delta step-down is worth noting.

	Titan to Atlas		Atlas to Delta				Delta to LLV3			
	GEO	HEO	GEO	MEO	LEO	HEO	GEO	MEO	LEO	HEO
Baseline (kg)	2254	1665	380	1262	3843	852	172	387	1890	365
Adv. Cryo	0.1		0.6	0.9	0.9	0.8	a	a	a	a
Nuc. Bimodal	0.4	1.0		0.5		0.3				
Solar Bimodal	0.4	1.1	0.8	0.9		0.9	1.5	1.6		1.3
Solar Thermal	0.5	1.2	1.3	1.0		0.9	3.6	2.1		1.9
Nuclear Ion	0.4			0.3						
Solar H_2 Arcjet	0.3		0.7	0.9	0.5		0.7	1.3	0.7	
Solar SPT	0.6		1.6	1.1	0.5		1.7	1.6	0.8	
Solar Ion	0.8		2.6	1.2	0.4		3.0	1.9	0.7	

■ >0.5 x baseline   [#] <0.5 step-down   ■ No step-down possible   ▒ Technology Not Suitable

[a]No LLV3 cryo lift design for comparison

Figure 5-19. Summary of the multiple of the baseline payload mass that can be stepped down to a smaller launch vehicle.

5.5.2.3 MOP 3-2: CAPABILITY OF HYPOTHETICAL NEW LAUNCH VEHICLE

Step-down potential assesses the capabilities of the innovative technologies when used in conjunction with existing launch vehicles. MOP 3-2 addresses an equally interesting question: If hypothetical launch vehicles were combined with the innovative technologies, what lift capabilities would they need to duplicate the existing baseline lift?

OCEM generates the results for this MOP using a top-down approach. Based on the satellite characteristics of Table 5-2 and the innovative propulsion and electrical power technologies, OCEM determines the launch vehicle drop-off mass for a specific drop-off orbit needed to place the complete representative payload into mission orbit. This is the mass a future launch vehicle would need to lift into the drop-off orbit assuming the innovative technology is used for the upper stage. Estimates of payload fairing requirements are shown on the table as well. The results from this analysis are presented in Table 5-14 through Table 5-17 for ORM 1 (GEO), ORM 2a (MEO-GPS), ORM 3a (LEO-Polar), and ORM 4 (HEO), respectively.

The step-down is indicated at the top of Tables 5-14–5-17, as is the definition of the representative payload and maneuver characteristics from Table 5-2. Below this information are other types of data: the throw weight in kilograms of the hypothetical new launch vehicle needed to duplicate the baseline performance; the corresponding capability of the current launch vehicle to deliver mass to the same drop-off conditions; the fairing volume estimate for the innovative technology based on the largest current fairings; and the relevant perigee-by-apogee altitudes of the drop-off orbit (inclination is chosen to minimize plane change with the innovative lift technology [see Section 4.9]).

Results are presented for the best instance of each lift technology. As anticipated from the step-down potential results, the throw weight of the hypothetical new launch vehicle typically is less—often substantially less—than the corresponding baseline throw weight. Exceptions occur for the nuclear electric and advanced cryogenic technologies. The difficulties of the nuclear electric technologies are directly related to the mass of the reactor. The explanation for the advanced cryogenic is given in the next paragraph. With the exception of the advanced cryogenic, the throw weight of the hypothetical new launch vehicle corresponds to a zero margin for the launch vehicle.

We have a unique situation with the advanced cryogenic resulting from the replacement of one cryogenic upper stage with another cryogenic upper stage. For example, in Table 5-14 the advanced cryogenic upper stage directly replaces the current Titan or Atlas Centaur cryogenic stage and uses the same orbital transfer strategy. Since the drop-off conditions are the same for both upper stages and

Table 5-14. ORM 1 (GEO): Throw Weights Required of Hypothetical Launch Vehicle to Allow 100% Step-down from Baseline Launch Vehicles

		Step-down from Titan IV				Step-down from Atlas IIAS			
		Payload 2254 kg Payload Power 5.0 kW EOL ΔV 51.6 m/sec MMD 10 yrs				Payload 380 kg Payload Power 3.0 kW EOL ΔV 51.6 m/sec MMD 10 yrs			
No.	Technology	Throw Weight (kg)	Titan IV Throw Weight (kg)	Fairing Volume[a] (m^3)	Perigee × Apogee (km)	Throw Weight (kg)	Atlas IIAS Throw Weight (kg)	Fairing Volume[a] (m^3)	Perigee × Apogee (km)
1-3	Advanced Cryo	5215	5215	65	GEO	3696	3696	25	0×GTO
1-5	Nuclear Bimodal	13340	21946	214	circular 185×185	6408	8635	89	185×185
1-8	Solar Bimodal	17673	21946	240	185×185	6453	8635	94	185×185
1-11	Solar Thermal	12245	21946	162	185×185	4575	8635	61	185×185
1-13	Nuclear NH_3 Arcjet	21252	17883	198	370×370	11533	7839	81	370×370
1-15	Nuclear H_2 Arcjet	17355	17883	290	370×370	9543	7839	132	370×370
1-17	Nuclear Xe SPT	13790	17883	176	370×370	7486	7839	69	370×370
1-19	Nuclear Xe Ion	10831	17883	182	370×370	5979	7839	71	370×370
1-21	Solar NH_3 Arcjet	16976	17883	96	370×370	6351	7839	35	370×370
1-23	Solar H_2 Arcjet	13624	17883	167	370×370	5214	7839	63	370×370
1-25	Solar Xe SPT	11021	17883	78	370×370	4112	7839	28	370×370
1-27	Solar Xe Ion	8776	17883	75	370×370	3266	7839	27	370×370

[a]Volume based on original launch vehicle fairing.

Table 5-14. (concluded)

No.	Technology	Step-down from Delta II Payload 2254 kg Payload Power 5.0 kW EOL ΔV 51.6 m/s MMD 10 yr			
		Throw Weight (kg)	Delta Throw Weight (kg)	Fairing Volume[a] (m^3)	Perigee × Apogee (km)
1-3	Advanced Cryo	1819	1819	12	0×GTO
1-5	Nuclear Bimodal	b	b	b	b
1-8	Solar Bimodal	3626	4991	46	185×185
1-11	Solar Thermal	2411	4991	31	185×185
1-13	Nuclear NH_3 Arcjet	8972	4787	65	370×370
1-15	Nuclear H_2 Arcjet	7375	4787	104	370×370
1-17	Nuclear Xe SPT	5993	4787	57	370×370
1-19	Nuclear Xe Ion	4910	4787	58	370×370
1-21	Solar NH_3 Arcjet	3432	4787	18	370×370
1-23	Soalr H_2 Arcjet	2938	4787	35	370×370
1-25	Solar Xe SPT	2286	4787	15	370×370
1-27	Solar Xe Ion	1898	4787	15	370×370

[a]Volume based on original launch vehicle fairing.
[b]Nuclear bimodal does not fit on Delta II.

since the Centaur places exactly the representative payload into mission orbit, the advanced cryogenic upper stage need only duplicate the Centaur performance. However, because the advanced cryogenic is more capable than Centaur, it could place more payload mass in mission orbit—something not being measured in this MOP. This is equivalent to saying there is a launch vehicle margin for these cases.

The results shown in Table 5-14 for step-down from Titan for ORM 1 (GEO) are possibly the most interesting, as the results for MOP 3-2 indicated that none of the innovative technologies allowed complete step-down to Atlas. Significant throw weight reductions are achievable for all the electric technologies, and the nuclear bimodal and nuclear Xe technologies. This is a clear indication that while Titan to GEO cannot be replaced using any of the OECS innovative technologies on Atlas, these same technologies can substantially reduce the size of any new heavy launch vehicle designed to perform like today's Titan IV.

Table 5-15. ORM 2a (GPS): Throw Weights Required of Hypothetical Launch Vehicle to Allow 100% Step-down from Baseline Launch Vehicles

		Step-down from Atlas IIAS				Step-down from Delta II			
		Payload 1262 kg Payload Power 1.0 kW EOL ΔV 80.6 m/s MMD 10 yr				Payload 387 kg Payload Power 1.0 kW EOL ΔV 80.6 m/s MMD 10 yr			
No.	Technology	Throw Weight (kg)	Atlas IIAS Throw Weight (kg)	Fairing Volume[a] (m^3)	Perigee × Apogee (km)	Throw Weight (kg)	Delta II Throw Weight (kg)	Fairing Volume[a] (m^3)	Perigee × Apogee (km)
2-2	Advanced Cryo	3855	3855	30	0×GPS TO	1875	1898	14	0×GPS TO
2-3	Nuclear Bimodal	6116	8182	85	185×185	[b]	4719	[b]	[b]
2-4	Solar Bimodal	5822	8182	83	185×185	2998	4719	43	185×185
2-5	Solar Thermal	4682	8182	62	185×185	2432	4719	31	185×185
2-6	Nuclear NH_3 Arcjet	8776	7385	77	370×370	5814	4515	55	370×370
2-7	Nuclear H_2 Arcjet	7651	7385	106	370×370	5162	4515	75	370×370
2-8	Nuclear Xe SPT	7231	7385	77	370×370	4957	4515	55	370×370
2-9	Nuclear Xe Ion	6628	7385	84	370×370	4655	4515	59	370×370
2-10	Solar NH_3 Arcjet	5319	7385	38	370×370	2661	4515	18	370×370
2-11	Solar H_2 Arcjet	4732	7385	56	370×370	2474	4515	29	370×370
2-12	Solar Xe SPT	4332	7385	36	370×370	2147	4515	17	370×370
2-13	Solar Xe Ion	3942	7385	36	370×370	1995	4515	18	370×370

[a]Volume based on original launch vehicle fairing.
[b]Nuclear bimodal does not fit on Delta II.

Table 5-16. ORM 3a (LEO-Polar): Throw Weights Required of Hypothetical Launch Vehicle to Allow 100% Step-down from Baseline Launch Vehicles

		Step-down from Atlas IIAS				Step-down from Delta II			
		Payload 3843 kg Payload Power 2.0 kW EOL ΔV 120 m/s MMD 7 yr				Payload 1890 kg Payload Power 1.3kW EOL ΔV 20 m/s MMD 5.5 yr			
No.	Technology	Throw Weight (kg)	Atlas IIAS Throw Weight (kg)	Fairing Volume[a] (m^3)	Perigee × Apogee (km)	Throw Weight (kg)	Delta II Throw Weight (kg)	Fairing Volume[a] (m^3)	Perigee × Apogee (km)
3-2	Direct	6313	5805	79	850×850	3223	3175	40	850×850
3-3	Cryo	6735	6735	80	185×850	3447	3447	40	185×850
3-5	Solar NH_3 Arcjet	6385	6508	76	370×370	3407	3537	39	370×370
3-6	Solar H_2 Arcjet	6456	6508	79	370×370	3494	3537	42	370×370
3-7	Solar N_2H_4 Arcjet	6448	6508	75	370×370	3467	3537	39	370×370
3-8	Solar SPT	6413	6508	77	370×370	3399	3537	40	370×370
3-9	Solar Xe Ion	6439	6508	77	370×370	3435	3537	40	370×370

[a]Volume based on original launch vehicle fairing.

Table 5-17. ORM 4 (HEO): Throw Weights Required of Hypothetical Launch Vehicle to Allow 100% Step-down from Baseline Launch Vehicles

		Step-down from Titan IV				Step-down from Atlas IIAS			
		Payload 1665 kg Payload Power 2.8 kW EOL ΔV 190 m/sec MMD 7.5 yrs				Payload 852 kg Payload Power 2.8 kW EOL ΔV 190 m/sec MMD 7.5 yrs			
No.	**Technology**	**Throw Weight (kg)**	**Titan IV Throw Weight (kg)**	**Fairing Volume[a] (m^3)**	**Perigee × Apogee (km)**	**Throw Weight (kg)**	**Atlas IIAS Throw Weight (kg)**	**Fairing Volume[a] (m^3)**	**Perigee × Apogee (km)**
4-2	Direct	3988	4092	48	185×39,464	2601	2721	32	1000×39,464
4-3	Cryo	[b]	[b]	[b]	[b]	2601	3175	32	1000×39,464
4-5	Nuclear Bimodal	7934	19134	142	185×185	5826	7728	81	185×185
4-7	Solar Bimodal	9549	19134	137	185×185	5224	7728	75	185×185
4-9	Solar Thermal	6806	19134	90	185×185	4622	7728	61	185×185

[a]Volume based on original launch vehicle fairing.
[b]Cryo is not used with Titan IV; step-down with this technology is not applicable.
[c]Nuclear bimodal does not fit on Delta II.

Table 5-17. (concluded)

No.	Technology	Step-down from Delta II Payload 365 kg Payload Power 1.0 kW EOL ΔV 95 m/s MMD 5 yr			
		Throw Weight (kg)	Delta II Throw Weight (kg)	Fairing Volume[a] (m^3)	Perigee × Apogee (km)
4-2	Direct	1261	1254	15	1000×39,464
4-3	Cryo	1261	2890	15	1000×39,464
4-5	Nuclear Bimodal	c	c	c	c
4-7	Solar Bimodal	2858	4447	40	185×185
4-9	Solar Thermal	2398	4447	31	185×185

[a]Volume based on original launch vehicle fairing.
[b]Cryo is not used with Titan IV; step-down with this technology is not applicable.
[c]Nuclear bimodal does not fit on Delta II

5.5.2.4 SUMMARY OF REQUIREMENTS FOR HYPOTHETICAL NEW LAUNCH VEHICLE

The results of Table 5-14 through Table 5-17 are summarized in Figure 5-20. The format of this summary is again similar to previous effectiveness summaries. The baseline launch vehicles and ORMs are given at the top of the figure with technologies at the left. Again, only the best nuclear electric technology and the three best solar electric technologies appear in the summary.

Numbers in the boxes again indicate MOP effectiveness, this time giving the ratio

$$\frac{\text{required throw weight of hypothetical new launch vehicle}}{\text{comparable throw weight of baseline vehicle}}$$

For this ratio, smaller numbers represent better performance, which is a change from previous summaries. One caution is required when interpreting the above ratios: each comparison represents only one point of comparison and therefore is only suggestive of an achievable reduction in throw weight.

Empty, darkly shaded boxes indicate the need for greater throw weight than the baseline provides. The empty, lighter-shaded boxes once again indicate unsuitable technologies. The advanced cryogenic ratio is everywhere 1.0 in accordance with the explanation given above.

	Titan IV		Atlas IIAS				Delta II			
	GEO	HEO	GEO	MEO	LEO	HEO	GEO	MEO	LEO	HEO
Adv. Cryo	1.0		1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0
Nuc. Bimodal	.61	.41	.74	.75		.75				
Solar Bimodal	.81	.50	.75	.71		.68	.73	.64		.64
Solar Thermal	.56	.36	.53	.57		.60	.48	.52		.54
Nuclear Ion	.61		.76	.90						
Solar H_2 Arcjet	.76		.67	.64	.99		.61	.55	.99	
Solar SPT	.62		.52	.59	.99		.48	.48	.96	
Solar Ion	.49		.42	.53	.99		.40	.44	.97	

<0.75 x current LV throw weight

0.75–1.0 x current LV throw weight

Worse than Baseline

Technology Not Suitable

Figure 5-20. Summary of fraction of baseline throw weight required for hypothetical new launch vehicle.

As would be expected from all previous results, the solar technologies are consistently the best performers regardless of baseline launch vehicle or ORM. Nuclear bimodal does well with respect to Titan and adequately with respect to Atlas. No innovative technology provides a significant advantage for the OECS LEO-Polar ORM.

6. COST ANALYSIS

6.1 INTRODUCTION

Analyzing the cost of developing a technology and incorporating it into a system is not easy. While it is partly a matter of extrapolating from past cost data and using hard-core engineering cost figures, it is largely a matter of judgment based on experience. Single-system point estimates of the cost of future systems tend to be unreliable. *Relative* point estimates of competing concepts are more credible for a number of reasons. First, since competing concepts typically share many elements in common, the estimation errors in those elements are shared as well, thereby diminishing their importance. Second, if the costs of all the concepts are determined under the same ground rules, then the methodology will be consistent. Bias—be it optimistic, pessimistic, or neutral—tends to affect all concepts similarly. OECS cost estimates are relative estimates. They were produced by a single cost group under one set of operating rules. Because operations and support (O&S) costs of the satellite payload and bus are not included, OECS cost estimates are not life cycle cost (LCC) estimates. Instead, OECS costs are expressed as *acquisition costs*, a category that includes all development and procurement costs. The decision not to include O&S costs was made for two reasons:

- O&S costs associated with the central issues of the study, on-orbit power and propulsion, are small and approximately equal for all OECS technologies because the missions are similar
- Payload O&S costs are highly dependent on satellite mission, and the nature of a particular mission is not relevant to OECS cost comparisons

In this chapter, we discuss the OECS cost-estimating process and present a summary of results from 15 reference cases representing a comprehensive cross-section of OECS technologies. The total cost of each reference case is dependent upon the cost of several different technology development processes and the cost of implementing propulsion and electrical power for the relevant technology. Elements common to each technology are launch vehicles, remaining subsystems of the spacecraft, and payloads. Additionally we will estimate total costs for each ORM for technology combinations covering a wide range of values for payload mass, payload EOL electrical power, and number of satellites launched. These are the basic data on which the cost-effectiveness analysis is based.

6.2 COST METHODOLOGY

Estimates of technology development and theoretical first unit (TFU) costs constitute the core of the cost analysis. Cost models for each are composed of cost estimating relationships (CERs) for each design component (i.e., for each work breakdown structure [WBS] element). There are approximately 45 CERs each for development and TFU costs.

The Rocketdyne Division of Rockwell International provided most of the CERs for these models, with some specialized contributions from The Aerospace Corporation. The individual CERs and their sources appear as Appendix F, "OECS Cost Estimating Relationships," which is available on computer disk to eligible users.

Figure 6-1 summarizes the OECS cost methodology. The left side of the figure shows technologies as inputs. The right side of the figure identifies principal cost outputs: estimates of the total acquisition cost of each technology to feed the cost-effectiveness analysis and estimates of cost uncertainty to allow cost-risk assessment.

A key task in the methodology is characterizing the elements of the WBS using size and/or complexity descriptors (e.g., mass, volume, electrical power [see Section 6.2.5]) that can serve as a basis for developing CERs. This is accompanied by methodology development to transform the WBS into a cost breakdown structure (CBS). Combining the results of these tasks with the costing ground rules and assumptions (Section 6.2.3) makes it possible to estimate development costs, production costs, and other acquisition costs. The makeup of these categories is described below.

Fifteen, detailed cost-reference cases were developed to represent all the relevant combinations of power and propulsion technology. Table 6-1 summarizes the combinations used for the reference cases. Each case corresponds to a combination of technologies for prime electrical power, orbital transfer, and major station keeping ΔV expenditures. Inexpensive, state-of-the-art hydrazine thrusters are included for minor station keeping requirements, such as east-west station keeping for GEO satellites. The information in the spreadsheets allows development of cost models for any OECS technology combination.

6.2.1 COST CATEGORIES

OECS costs can be classified as development, production, or related costs. *Development costs* are "non-recurring (one-time) costs that move the technology from its current state to initial operational capability (IOC)." *Production costs* are "recurring manufacturing costs." *Related costs* include "anything that cannot easily fit into development or production, such as costs of facilities needed to support a given technology." The following section briefly describes relevant costs in each of these three categories.

6.2.1.1 DEVELOPMENT COSTS

Technology Acquisition Cost

Technology acquisition cost accounts for activities preceding design, development, test & evaluation (DDT&E) activities. It represents the cost incurred to bring power and propulsion technologies to the NASA technology readiness level (TRL) assumed by the

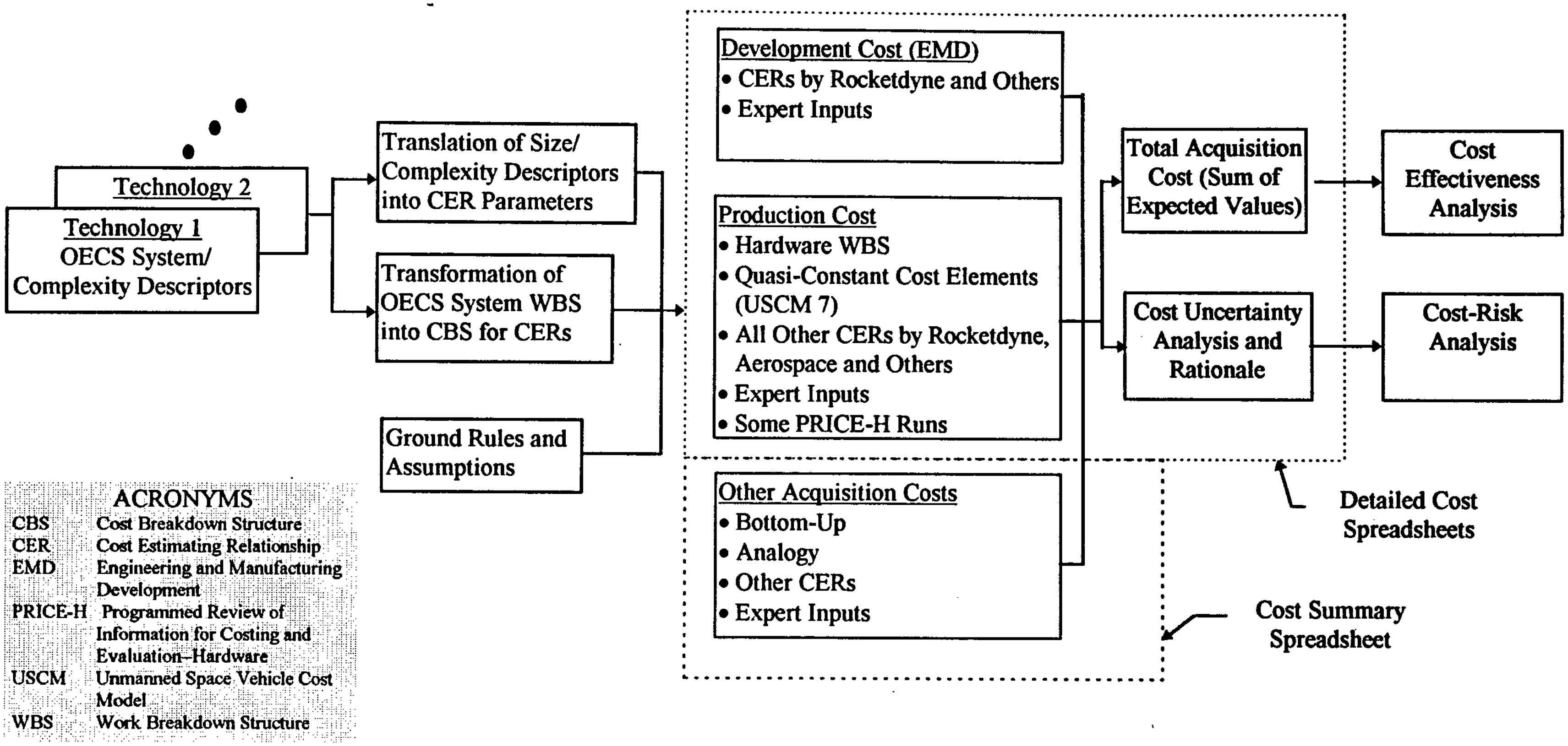


Figure 6-1. OECS costing process.

Table 6-1. Technology Combinations Used in the Cost Spreadsheets

Reference Case Number	Orbital Transfer Technology	Station-keeping Technology	Electric Power
1	Chemical Baseline	N_2H_4 Arcjet	Photovoltaic
2	Solar Electric (H_2 Arcjet)	N_2H_4 Arcjet	Photovoltaic
3	Solar Electric (NH_3 Arcjet)	NH_3 Arcjet	Photovoltaic
4	Solar Electric (Xenon Ion)	Xenon Ion	Photovoltaic
5	Solar Electric (Xenon SPT-100)	SPT-100	Photovoltaic
6	Solar Thermal (H2)	Photovoltaic N_2H_4 Arcjet	Photovoltaic
7	Solar Bimodal (H2)	N_2H_4 Arcjet	Solar Thermionic
8	Nuclear Electric (H_2 Arcjet)	SPT-100	Nuclear Incore Thermionic
9	Nuclear Electric (NH_3 Arcjet)	NH_3 Arcjet	Nuclear Incore Thermionic
10	Nuclear Electric (Xenon Ion)	Xenon Ion	Nuclear Incore Thermionic
11	Nuclear Electric (Xenon SPT-100)	SPT-100	Nuclear Incore Thermionic
12	Nuclear Bimodal (Concept 3) (H2)	N_2H_4 Arcjet	Excore Thermoelectric
13	Chemical w/new Delta 7925 Cryo Stage	N_2H_4 Arcjet	Photovoltaic
14	Chemical w/new Atlas IIAS Cryo Stage	N_2H_4 Arcjet	Photovoltaic
15	Chemical w/new Titan IV Cryo Stage	N_2H_4 Arcjet	Photovoltaic

CER, typically between TRL 5 and TRL 6. The NASA TRL scale is described in Figure 6-2. Technology acquisition cost includes ground testing but not flight demonstration cost. Estimated acquisition costs for the proposed OECS technology options are as follows:

Power Technology

- Photovoltaic: $29M
- Solar bimodal (SEBA 2): $16M
- Nuclear static electric: $35M
- Nuclear bimodal (NEBA 1): $37M

Propulsion Technology

- Biprop (chemical) or mono-prop: $0
- Cryo: $0
- Cryo, IME stage: $50M
- Solar thermal: cost included in solar bimodal power technology (see above)
- Arcjets: $10M
- Advanced SPT: $10M
- Ion: $10M

The rationale for these estimates is based on Rocketdyne development schedules, Phillips Lab estimates, and design and testing of critical components.

Design, Development, Test, and Engineering (DDT&E) Cost

DDT&E cost is frequently referred to as development cost. It represents the cost incurred to bring power and propulsion concepts from TRL 6 to initial operational capability (IOC) at TRL 9. Formerly referred to as full scale development (FSD), the phase represented by this cost is now named engineering and manufacturing development (EMD).

DDT&E cost encompasses system design, prototype production, aerospace safety, ground testing, and qualification testing of the systems. It excludes flight demonstration costs. Aerospace safety cost (some reactor components, mockup hardware and nonnuclear test articles) is included in DDT&E for nuclear cases. Representative DDT&E costs range from approximately $100M to $900M.

Flight Demonstration Cost

Flight demonstration cost is the cost of demonstrating the technology in space. For the solar bimodal, solar thermal, and solar electric designs, we use one half of a two-module flight prototype and fly on the smallest capable launch vehicle. For all other concepts, we have assumed a full-up flight prototype flown on the smallest capable launch vehicle. In all cases, flight demonstrations are assumed flown without payloads. Total flight demonstration cost consists of the costs of the following elements:

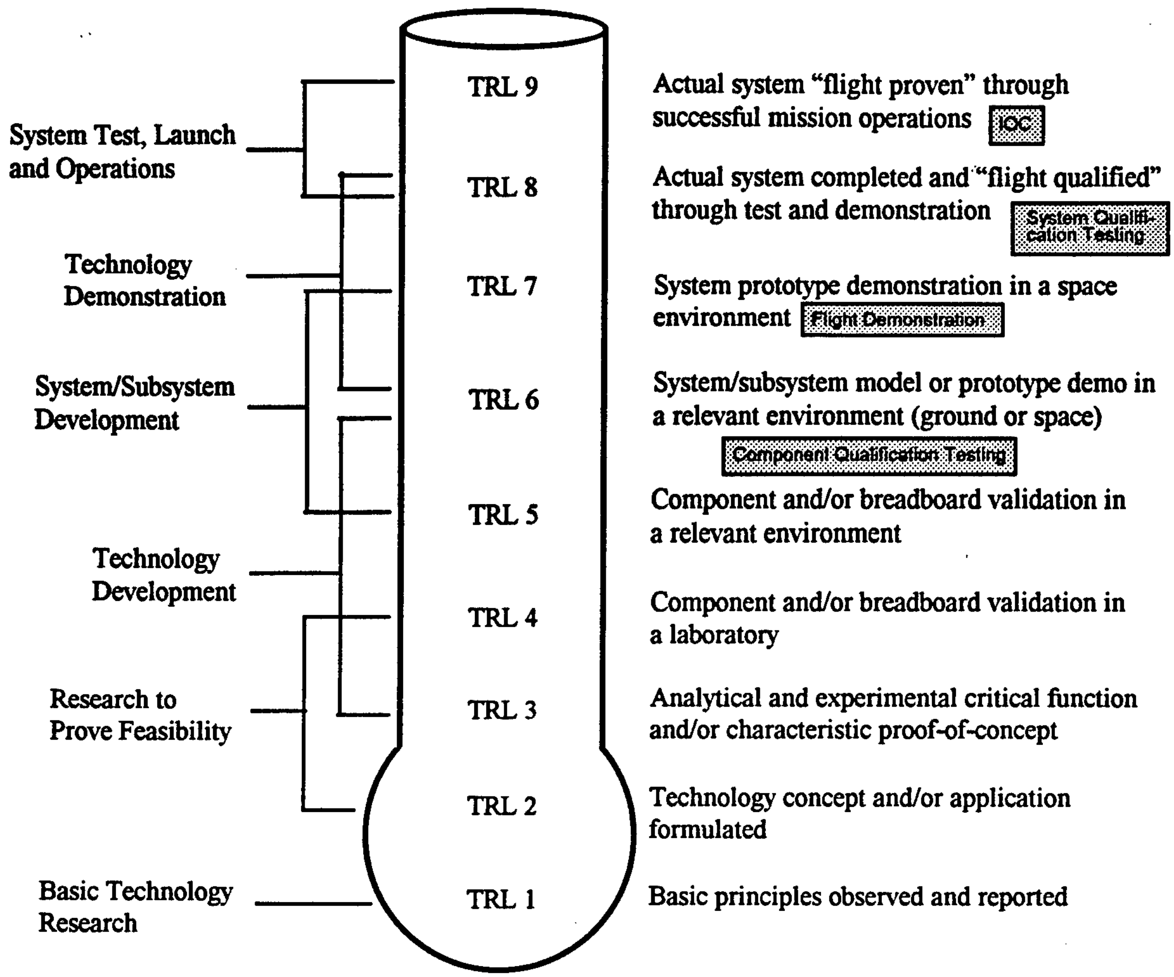


Figure 6-2. NASA technology-readiness-level (TRL) scale.

- Flight prototype unit
- Launch vehicle
- Flight support
- Ground software

For most designs, the most expensive part of a flight demonstration is launch cost. OECS flight demonstration cost estimates range from $34M to $239M. Details are given in Table 6-6 in Section 6.2.6.

6.2.1.2 PRODUCTION COSTS

Production costs are expressed in terms of subassembly TFU costs. TFU production cost of the lowest CER-level subassembly (e.g. thruster, battery, etc.), includes manufacturing engineering support. TFU costs for the 15 spreadsheets range from $30M to $86M.

The term *spacecraft* is used here to mean "the bus but not the payload." Figure 6-3 shows the spacecraft bus WBS used to develop hardware CERs. Spacecraft subsystems are discussed in more detail in Section 6.2.4.

Some OECS upper-stage concepts use high I_{sp}, low-thrust propulsion leading to long orbital transfer times. To ensure a given constellation availability when faced with long transfer times, on-orbit spare satellites must be maintained, i.e., more satellites must be launched if these concepts are used than in the case of short transfer times (see Sections 4.2.1 and 4.3.1). Because the number of satellites required for different concepts varies, payload costs must be considered in addition to upper-stage costs to assure a fair cost comparison. OECS payload costs of various generic types are determined from payload CERs used in the Aerospace Satellite Cost Model (Hovden and Dickey). Separate CERs for surveillance, communications, and weather/navigation payloads are each evaluated based on payload weight.

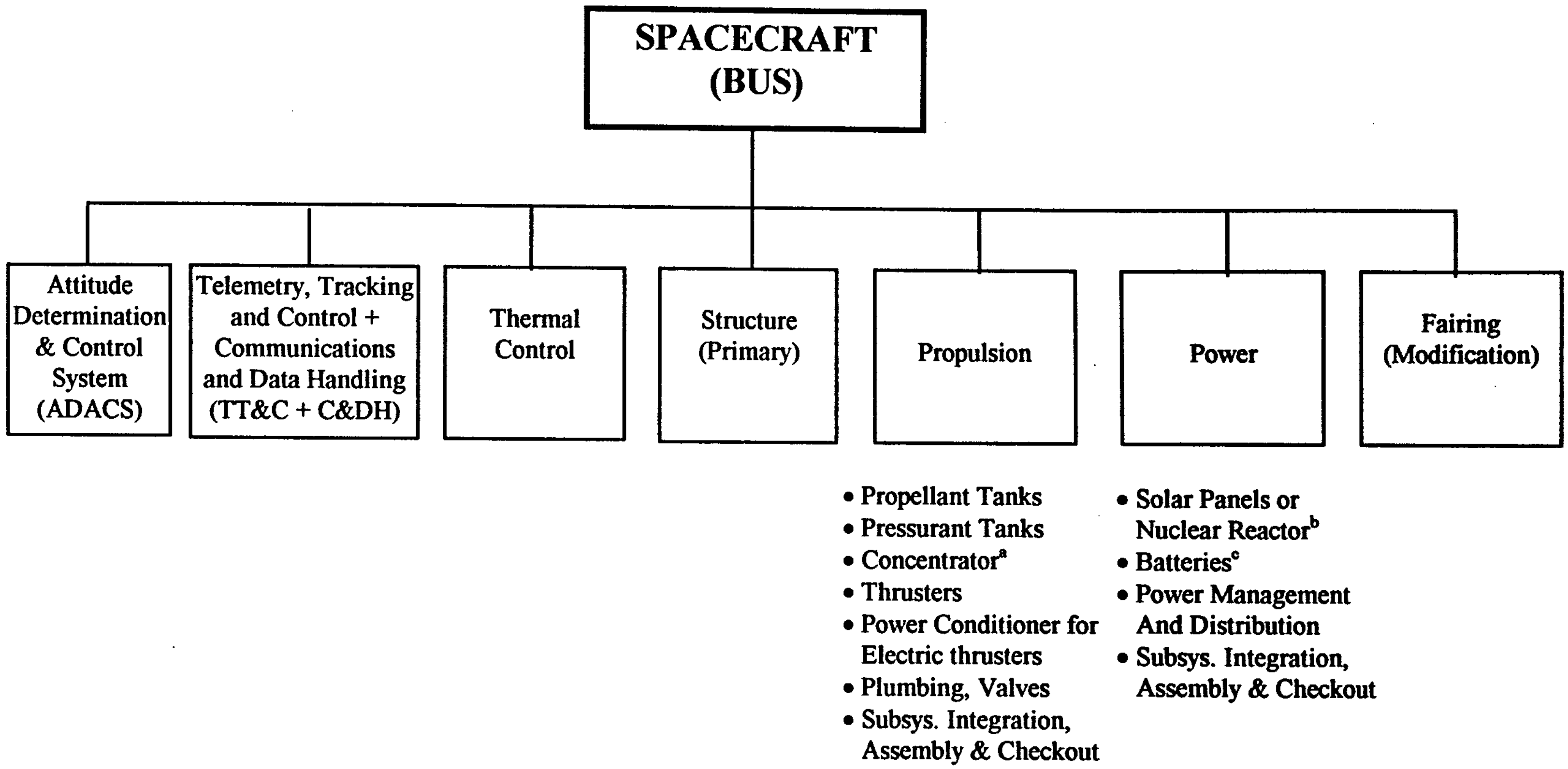
6.2.1.3 OTHER ACQUISITION COSTS

Facility costs are incurred for new test facilities, modification of existing test facilities, and new or modified major ground support equipment (e.g., a shipping cask for reactors). Also included in this category are new or modified production facilities if they are required for government spacecraft (e.g., photovoltaic cell production) and new or modified launch facilities. Facility costs vary from a low of $25M to a high of $225M depending upon technology.

The cost of nuclear technology facilities may be reduced by eliminating full-power ground testing. Initially the SP-100 program (the predecessor of NEBA-3) called for a full-power ground test, but this test was dropped in favor of a space-based full-power test. Very-low-power ground tests of the reactor and full-power nonnuclear tests of the hardware were used to qualify the components. Full-power ground testing drives some of the nuclear facility costs and the aerospace safety costs. Some ground power testing at full or reduced power appears to be needed for the following reasons:

- Upper stages with very high reliability requirements (those that require ground testing at higher than operating power to demonstrate adequate margins)
- Complexity of the system (especially nuclear bimodal systems)
- Potential for future work on anomaly resolution and performance upgrades
- Long-life testing of the entire power system

However we assume the development program of a nuclear system requires only two, equivalent, complete power/propulsion systems for system and component ground testing and qualification. This differs from nonnuclear systems, where more than two equivalent systems are often ground tested prior to first flight. The ground testing of nuclear bimodal systems primarily supports qualification and certification of power life. This is why we retained full-power ground testing of nuclear technologies.



[a]For solar thermal propulsion only.
[b]Nuclear reactor subsystem: nuclear reactor proper, power converter, radiation shielding, boom structure, radiator, and thermal control.
[c]For photovoltaic systems only.

Figure 6-3. Spacecraft subsystems and associated work breakdown structure for propulsion and power.

The nuclear bimodal costs (Case 12) are based on NEBA 1. Martin Marietta (Brown) and DOE gave us an alternate design—NEBA-3—and cost estimates based on the SP-100 after the contract for cost analysis had expired. It is not clear at this point whether NEBA 3 would provide significant savings over NEBA 1.

Launch costs are also included in the category of other acquisition costs. The study assumes launch on an LLV3, Delta 7920/7925, Atlas IIAS, or Titan IV. The OCEM sizing model determines the smallest vehicle on which the payload and upper stage can be launched based on the combined satellite and upper-stage masses (see Chapter 4). The Aerospace Corporation has assigned launch costs to all the launch vehicles. One launch vehicle is required for each satellite launched. Table 6-2 summarizes per launch costs.

Table 6-2. OECS Costs per Launch (FY95 $M)

Launch Vehicle	Cost per Launch[a]	Upper Stage/Payload/ Booster Integration
LLV3	$26	$5
Delta 7925	$58	$10
Delta 7925 with new cryogenic stage	$67	$10
Atlas IIAS	$104	$10
Atlas IIAS with new cryogenic stage	$95	$10
Titan IV NUS (No upper stage)	$224	$10
Titan IV Centaur	$293	$10
Titan IV with new cryogenic stage	$242	$10

[a]Includes launch vehicle hardware, launch support, range operations, and other government costs.

All Titan IV configurations are assumed to use two Titan Solid Rocket Motors–Upgraded (SRMUs). Except for the LLV3, cost estimates for the above configurations are projections from actual program offices costs. Cryogenic upper-stage cost estimates are based upon several models and sources, including:

- Air Force Launch Vehicle Cost Model developed by Tecolote, Inc., in 1989
- NASCOM Model developed by NASA Marshall Space Flight Center in 1993
- PRICE-H Cost Model
- Centaur actual stage costs
- ICBM actual avionics costs modified for space launch
- *Operational Integrated Modular Engine Contract Final Report* (Rockwell International, 1992)

Facility costs, which include ground testing, fuel fabrication, storage and integration facilities, etc., were estimated by analyzing the requirements for development, production and launch operations. Cost estimates are based on Rocketdyne experience, NASA Lewis Research Center history, and Phillips Laboratory predictions. Facility costs are fairly soft estimates, since they depend on availability of existing assets, type of testing required, amount of use of non-US facilities, allocation schemes, and other factors. Ground support equipment (GSE) and special test equipment (STE) are included in facility costs. GSE and STE costs are normally separate cost items, but they are included here with facility costs for convenience.

The largest facility costs are for the nuclear electric and nuclear bimodal technologies. The cost estimate of $225M for nuclear electric facilities includes:

- $100M for nuclear ground-testing facility modification
- $40M for shipping cask development, production, and test and qualification
- $10M for launch-site storage and payload-integration facility
- $75M for GSE and STE

The cost estimate of $180M for bimodal reactor facilities includes:

- $20M for a closed-loop engine stand
- $10M for modifying the fuel lab
- $10M for an electrical furnace
- $40M for a shipping cask
- $10M for refurbishing the criticality facility
- $10M for refurbishing the launch-site storage facility
- $5M for modifying the payload-integration facility
- $75M for GSE and STE

The cost estimates for other technology facilities are:

- $6M for a new photovoltaic production facility (paid for by the ManTech program)
- $25M for photovoltaic GSE and STE
- $60M for solar thermal and solar bimodal
- $75M for the advanced cryogenic stage

The rationale for the dollar values of GSE and STE is as follows. A recent detailed logistics and support analysis for a generic single-stage-to-orbit (SSTO) booster engine identified 32 types of handling equipment, test equipment, maintenance equipment, and protection equipment for rocket engines. Engineering experts estimated the design hours, support hours, and material costs at about $25M. The following subjective cost factors were applied to the original $25M: 1.0 for photovoltaic power/propulsion systems; 2.0 for solar thermal and solar bimodal power/propulsion systems; and 3.0 for advanced cryogenic propulsion systems and nuclear and nuclear bimodal power/propulsion systems. These factors were based on the complexities of the systems (stage versus a stand-alone cluster of booster engines); propulsion and power functions of the bimodal concept; number of test, production and launch sites where equipment will be needed; and modularity of the systems.

"Other costs" also include the estimated cost of launch-vehicle fairing modification. While fairing modifications may not be required, the fairing may occasionally need to be lengthened to accommodate the cryogenic hydrogen tank. This could be the case for any of the thermal technologies (nuclear bimodal, solar bimodal, and solar thermal) and nuclear and solar electric designs using H_2. The need for fairing modification will depend on mission requirements. Fairing volume constraints are discussed in Chapter 5. For simplicity, fairing modification costs have not been included in any OECS total costs. However, CERs for these modifications are given in Appendix F.

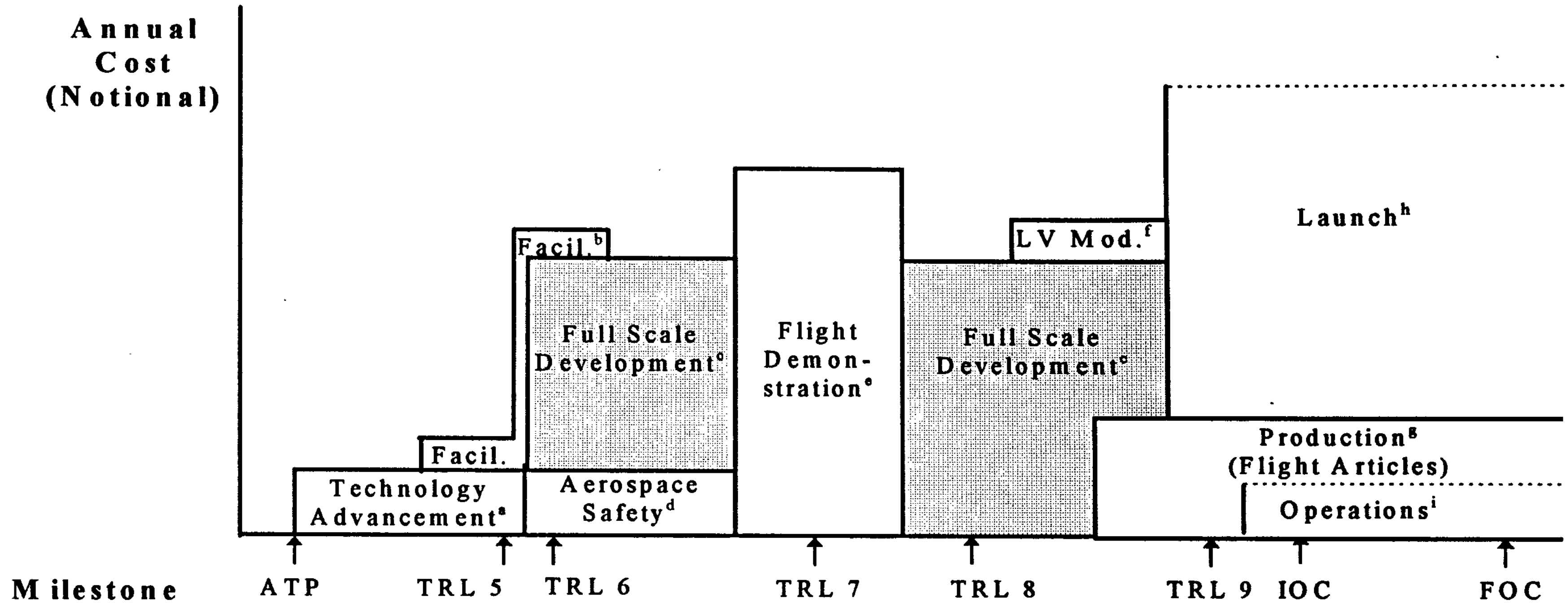
6.2.2 Program Spending Profile

Figure 6-4 displays the notional program spending profile used in the analysis. The profile is tied to milestones beginning with authority to proceed (ATP) and ending with full operational capability (FOC).

6.2.3 Cost Ground Rules and Assumptions

The cost panel developed the following cost ground rules and assumptions:

1. Only system acquisition and launch costs are considered (i.e.,operations and support costs are not considered).
2. All costs are presented in constant FY 1995 dollars.
3. All costs are contractor costs, including general and administrative (G&A) and procurement costs and subcontractor fees but excluding any integrating contractor fee.
4. No cost estimates are provided for government support for contingencies; however, launch costs are all inclusive
5. System production costs (upper-stage and satellite) are expressed as the cost of the TFU. No cost improvement is included for fewer than 11 satellites. Ninety percent learning is applied to the entire production cost of N items when more than 10 items are produced, that is, production cost = TFU $\times$ $[9 + (N - 9)^{0.848}]$.
6. Costs are determined as "most likely values." Seventy percent (high) and 30% (low) costs for the 15 representative cases are determined for cost risk analysis.
7. All systems are unmanned.
8. New-ways-of-doing-business factors are considered implicitly or explicitly.
9. A development cost factor assigns credit for past or ongoing development efforts.
10. The endpoint of the technology advancement phase is TRL 5.
11. CERs for developmental cost cover NASA technology readiness levels TRL 6 through 9, but exclude flight demonstration costs. Flight demonstration costs are accounted for separately.



[a]Technology acquisition or risk reduction
[b]Test and production facilities
[c]Design, development, test, and evaluation
[d]For nuclear systems only
[e]For all systems
[f]Launch vehicle modification (fairing)
[g]All flight hardware
[h]Launch vehicle and launch support
[i]Operations and flight support (excluded from OECS)

Figure 6-4. Notional spending profile for OECS program.

12. Pre-planned program improvement (P^3I) is not considered due to insufficient planning.
13. All technologies and launch vehicles are assumed to have the same reliability—which is a study ground rule as well. (Impact of reliability on cost is considered in the cost-effectiveness analysis.)
14. Cost of satellite disposal is included in propulsion propellant requirements.

6.2.4 Elements of the Spacecraft Subsystems

Figure 6-3 shows the hardware WBS used for CERs. The spacecraft, often called the bus or platform, consists of six subsystems. (Fairing modification, which is also shown in Figure 6-3, is not a subsystem.) This section describes the principal functions of the six subsystems and indicates typical hardware components. Table 6-3 summarizes OECS independent parameters and sources.

6.2.4.1 ATTITUDE DETERMINATION AND CONTROL SYSTEM (ADCS)

The attitude-determination-and-control system encompasses all components used in sensing and determining the proper orbit and attitude for the space-vehicle. Sensor components include gyroscopes, electronics, magnetometers, earth sensors, sun sensors, and star trackers. The control system consists of components (i.e., magnetic torque's and reaction wheels) that restore or maintain proper attitude and orbit. The primary cost driver is mass. ADACS costs are similar for all cases. ADACS CERs are taken from the Air Force Unmanned Space Vehicle Cost Model, Version 7 (Nguyen)

6.2.4.2 TELEMETRY, TRACKING, AND COMMANDING PLUS COMMAND AND DATA HANDLING (TT&C + C&DH)

The TT&C and C&DH subsystem measures space vehicle platform conditions, processes them along with mission data, stores the data, transmits them to the ground, receives and processes commands from the ground, initiates their execution, and provides a tracking capability. Typical subsystem equipment includes analog-to-digital converters, coders, digital electronics, signal conditioners, transmitters, format control units, antennas, receivers, and decoders. The primary cost driver is mass. TT&C + C&DH costs are similar for all cases and are based on USCM 7 CERs.

6.2.4.3 THERMAL CONTROL

Thermal control maintains the temperature of the spacecraft and mission equipment by modifying heat transfer to and from each element so temperature remains within allowable ranges throughout the mission. In general, unmanned space vehicles use passive temperature control. The primary cost driver for the thermal control subsystem is mass.

Table 6-3. Typical CER Characteristics

Subsystem/ Component	Independent Parameters	Comments	CER Source
ADACS	Mass	Similar for all cases	USCM 7
TT&C	Mass	Similar for all cases	USCM 7
C&DH	Mass	Similar for all cases	USCM 7
STRUCTURE			
Primary	Mass	Complex, mechanisms separately included	USCM 7
Radiation shielding	—	Included in PMAD	USCM 7
PROPULSION			
Propellant tanks	Volume, fluid type		T17, vendor data
Pressurant tanks	Volume		T17, vendor data
Thrusters			
Chemical	Thrust, complexity		T17, RD data
Resistojet	Thrust, complexity		RD data
Arcjet	Thrust, complexity		RD data, vendor data
Ion	Thrust, complexity		RD data
Solar	Thrust, complexity	Analogy with chemical	Solar Thermal Propulsion Transfer Stage Study (STPTS)
Thrust power condition	kWe		T21
Plumbing	—	Factor on propulsion subsystem	Propulsion engineering estimate
POWER			
Photovoltaic			
Solar cells	BOL kWe, solar cell type		T21
Batteries	kW hr, battery type		T21
Radiator	Hex area		T21
Thermal control	kWe		T21
PMAD	kWe, type		T21

Table 6-3. (concluded)

Subsystem/ Component	Independent Parameters	Comments	CER Source
POWER (concluded)			
Solar thermal			
Collector	kWth	Scaled from reflectors	T21, vendor
Absorber	kWe		T21
PMAD	kWe, type		T21
Nuclear			
Nuclear reactor proper with converter	kWth, reactor type		
Static (incore thermionic)	kWth, reactor type		T17
Static (excore thermionic)	kWth, reactor type		T17
Shielding	Mass		Expert input
Boom/structure	Mass		T17
Heat rejection	kWth, cycle, max temp.		T17
PMAD	kWe, type		T17
Propellant supply	Mass, complexity		T17
Controls/sensors/ software	Constant/source lines of code/platform		T17
STRUCTURE, PROPULSION & POWER SYSTEM FACTORS			
System assembly	Factor		T17, T21
System acceptance	Factor		T17, T21
Production management	Factor		T17, T21

6.2.4.4 STRUCTURE

Structure (primary) serves as the central frame of the space vehicle, which provides support and mounting surfaces for all equipment. The USCM 7 definition of structure also includes mechanisms, interstage, and all mechanical assemblies. Structure mass is the principal cost driver for this subsystem. For solar thermal and solar bimodal concepts, structure mass is divided into 80% static and 20% complex dynamic. (An example of complex dynamic is high precision bearings with 2 degrees of freedom.) The cost of complex dynamic structures was taken as two times the cost of fixed structures.

6.2.4.5 PROPULSION

The propulsion subsystem provides reaction force for maneuvers into orbit and for on-orbit changes. Typically, it consists of liquid rocket engines along with tankage, plumbing, thrusters, and power conditioners. Primary cost drivers are tank volume, thrust level, thruster complexity (integration, assembly and testing), and electrical power in kilowatts for thruster power conditioners. The OECS propulsion CERs (more than 50 of them) are taken from Rocketdyne's T17 (Meisl, March 1993) and T21 (Meisl, November 1993) documents.

6.2.4.6 ELECTRICAL POWER

The electrical power subsystem generates, converts, regulates stores and distributes all electrical energy to and between space vehicle components. Typical equipment includes solar cells or nuclear reactors, regulators, converters, power management and distribution (PMAD) units, batteries and wire harnesses. Primary cost drivers are mass, beginning of life power, and battery capacity. OECS power CERs are taken from Rocketdyne's T17 (Meisl, March 1993) and T21 (Meisl, November 1993) documents.

6.2.5 Cost Models and Sources

Rocketdyne Division of Rockwell International Corporation had primary responsibility for selecting and implementing the CERs and cost data bases for this study. Rocketdyne used several cost models and cost sources. The Aerospace Corporation provided cost projections and CERs for some items where Rocketdyne's information was incomplete, notably for launch vehicles and payloads. The Aerospace Corporation also performed spot cross-checks on Rocketdyne's estimates of spacecraft and cryogenic upper stages utilizing Aerospace Corporation's Satellite Cost Model and Launch Vehicle Cost Model, respectively.

The contractor's cost-estimating spreadsheets were linked to The Aerospace Corporation's OCEM model, creating an automated tool for analysis of the many technology/mission combinations addressed by the study. A variety of cost models and other sources were utilized to estimate costs of space segment hardware. The CERs and supporting references are contained in Appendix F.

The OECS cost model uses CERs to estimate TFU and DDT&E costs. Each configuration is broken down into a set of subsystems and components. The CERs estimate TFU and DDT&E costs for each of the components. These costs are totaled to produce overall TFU and DDT&E estimates of the configuration costs.

The simplest CERs are constants. A constant CER implies that the cost for this component is not a function of any size attribute, at least within the precision of our cost model for the size range used in the designs. The form of the CER would then be:

$$Cost = Constant \tag{6-1}$$

The most common form for the CERs used in the cost models is:

$$Cost = aX^b + c \tag{6-2}$$

where a, b, and c are constant coefficients and X is a size variable. The size variable describes dimensions, weight, power, thrust, or other determining characteristics of the subsystem or component. The valid range of the size variable for TFU cost is specified by the data used to create the CER. While no corrective action is taken if the CER range is exceeded in the OCEM model, the fact is noted for review. In general, CERs for production cost have two major parameters: size and complexity. For nuclear reactors, the size parameter is a minor cost driver up to a thermal power level of about 1 MW due to minimum size considerations for criticality. The production-cost CER assumes size independence of nuclear reactors up to 1 MW and minor size dependence (exponent of 0.2 on thermal power) up to power levels of about 2000 MW. (Beyond that, the CER uses gas-cooled reactor data with a power exponent of 0.6).

A more general CER can be derived from this equation by multiplying it by one or more additional factors, f_i, making the form of the CER

$$Cost = F(aX^b + c) \tag{6-3}$$

where $F = f_1 f_2 \ldots f_n$ and n is the number of applicable factors. The f_i are explained in Table 6-4.

A few CERs have more than one form, based on the value of the size variable. These CERs typically have the more complicated form of Equation 6-3 for the high end of their valid range and the simple form of a constant at the low end.

The CERs for the development cost generally assume the relevant technologies are already advanced to approximately TRL 6. In other words, the data on which the CERs are based come from programs of similar efforts with already proven technology that must be developed into a system and fielded. To use these data in estimating development costs where technologies are at different readiness levels, we have applied a multiplicative

Table 6-4. Explicit or Implicit Cost Factors in the Spreadsheets

Factor	Explanation	Application
Development cost	Adjusts development costs downward if technology readiness level (TRL) is initially greater than 6	f_i in Equation 3. Applied to design, analysis, and engineering part of development costs (but not hardware)
Escalation	NASA inflation index for escalation to FY 1995$	f_i in Equation 3
Material procurement expense (MPE) factor	Explicitly given as a CER modification, but implicitly incorporated in most CERs (unless stated otherwise)	f_i in Equation 3
Platform	Accounts for conversion of man-rated to unmanned where CER was derived from man-rated data (≈0.5)	f_i in Equation 3
Complexity	Accounts for complexities not contained in CERs	f_i in Equation 3
Miscellaneous hardware	Accounts for items (e.g., plumbing, valves, etc.) whose cost can be estimated as a fraction of another cost (generally 10%)	
Spacecraft integration	Accounts for total system integration of all subsystems. Covers work typically performed by systems integration contractor. Includes total system integration, assembly and checkout; acceptance or ground test; production management; SE & I; DDT&E management, integrated contractor G&A. Excludes contractor fee. Adjusted for new ways of doing business.	"Bottom line" systems cost factor. In OECS cost analysis, a factor was applied separately to the total cost of each sub-system, since the factor was different for different CER sources.

factor to the CER (see Ground Rule 9). The factor is less than 1.0 if the systems are already well known or commercially available and 1.0 otherwise (assumes TRL 6). The rationale for the development factor and the factor values are given in Table 6-5.

An escalation factor, the second factor in Table 6-4, inflates the CER result to FY1995 dollars, correcting for the different base year used in the formulation of the CER. NASA escalation factors were used instead of lower DOD factors because the technologies considered in OECS were developed mostly under NASA contracts or were extrapolations of NASA-developed hardware. The PRICE-H model used in some of the cases has an escalation factor that is quite close to the NASA factors.

Table 6-5. Rationale for Development Cost Factors

Subsystem/Component	F	Rationale
Radiator (heat pipes)	0.5	Considerable technology work has already been done (TRL 7-8)
PMAD (power management and distribution)	0.8	Components are at TRL 7-8, and power levels are comparable to ISSA components, but ISS has not flown yet
Power converter Power systems control/sensor/software	0.8	Some of work performed during SP-100 reactor program is assumed to be applicable (TRL 7)
Propellant tanks Electric thrusters	0.8	Similar subsystems are already flying, but substantial modifications are required
Structures	0.8	Similar structures are already flying, but adaptations and new interfaces need to be addressed
NiH_2 battery	0.1	Developed product, but requires technology improvements to obtain lower weights

The third factor in Table 6-4 accounts for material procurement expense (MPE) included in some CERs but not in others. CERs whose formulation does not include MPE are multiplied by an MPE factor of 1.12 (i.e., 12% material procurement expense) in all cases. Twelve percent is a significant reduction from historical MPEs of 25% or more. Savings are based on an assumed streamlining through a "new way of doing business" (NWODB). MPE factors reflect Rocketdyne experience.

The platform or unmanned factor adjusts CERs based on data for manned space programs since this project involves only unmanned systems. Unmanned adjustment factors range from a high of 1.0 (meaning that the data are, in fact, based on unmanned programs) to a low of 0.45 (Price-H models). The unmanned adjustment factor estimates cost savings to be realized if reliability and safety considerations of manned programs were not levied on the program.

The complexity factor typically corrects the CER for complexity when the CER is based on data that may be closely related to the subsystem or component in the design, but are not identical in scope or complexity.

A few component costs are estimated as a percentage of some specified subtotal. For example, we estimate costs of plumbing, valves, and miscellaneous hardware in the propulsion subsystem as 10% of the subtotal for all other components in the propulsion subsystem after multiplying by all other factors. We also add an integration, assembly, and check-out component to both the propulsion and power subsystems. We estimate this integration, assembly, and check-out component as 10% of the subtotal of all components in these subsystems, including the plumbing, valves, and miscellaneous line item.

One final factor, the integration factor, multiplies the costs of each major subsystem before the subsystems costs are totaled to produce the final cost for the configuration. This factor includes the following:

1. Total Spacecraft Integration, Assembly and Checkout (IA&T): The first grouping, called subsystem IA&T, addresses costs of integrating and assembling individual components into a subsystem. In USCM 7, subsystem IA&T costs are embedded in subsystem CER values. Rocketdyne CERs exclude subsystem IA&T costs, but include IA&T costs as a factor, usually 10% of production and DDT&E costs.

The 10 % factor derived from Rocketdyne's experience is applied to all subsystem development costs, including full-power nuclear ground tests, subsystem tests and subsystem nonhardware design and analysis activities. All these activities have to be integrated. The integration "charge" is not only for flight hardware, but also for development hardware and for nonhardware systems engineering activities (e.g., configuration control, interface consistency, reliability and safety engineering at the subsystem level).

2. Acceptance Test/Ground Test: This category includes ground support equipment (electrical and mechanical) required to support the space vehicle during ground test and preparation for flight operations. It is usually 10% of production and DDT&E costs.

3. Production Management: This category, usually 5% of production costs, includes all efforts associated with defining, planning, and directing to accomplish production objectives.

4. Systems Engineering and Integration (SE&I) and DDT&E Management: This includes all efforts associated with the engineering organization. Also included are costs associated with controlling system-level documents. It is usually 25% of DDT&E costs.

5. Integrating Contractor General and Administrative (G&A): This is usually 10% of production and DDT&E. The integrating contractor fee is excluded for USCM 7 and the Rocketdyne CERs.

The total spacecraft integration factor is a "tax" levied on the aggregated system cost for the total integration of all subsystems into a combined power/propulsion system, also known as "bus." It includes telemetry, attitude control, the structure holding all the subsystems together, etc. This factor, which incorporates the NWODB (a 50% reduction in all factors except G&A), is 0.89 for production and 0.84 for DDT&E for the USCM 7 CERs; and 1.25 for production and 1.39 for DDT&E for the Rocketdyne CERs. The spacecraft integration factor is applied at the bottom line to sum all subsystem production and development costs, respectively.

6.2.6 Flight Demonstration Costs

Table 6-6 gives our cost estimates for full-scale, on-orbit flight demonstrations for each of the 15 major technology pairs. These estimates are primarily functions of the TFU cost of the full-scale system and its weight. In all but the nuclear cases, cost of flight-demonstration hardware is assumed to be one-half the TFU cost of the full-scale system. Due to critical mass considerations, it was assumed that it was not feasible to launch part of the nuclear system (such as is possible for nonnuclear modular systems). Although a flight demonstration for a smaller, nonnuclear power plant could be envisioned, it would require separate flight qualification of the smaller power plant, which in turn might cancel cost advantages due to booster stepdown for the flight demonstration. In addition, some critical performance parameters, such as nuclear and thermal transients, may not be scalable by size. Because nuclear systems require a higher degree of certainty in their demonstration, we assume a full system must be flown and tested as a demonstration unless later, more detailed analyses come to a different conclusion. Thus, the cost of flight hardware for nuclear systems is assumed to be equal to the TFU cost.

The cost of launching demonstration hardware is also a significant portion of flight-demonstration cost. We assume the hardware is launched on the least expensive launch vehicle capable of placing it in the desired orbit. Launch costs are driven by the launch vehicle costs, which are determined by hardware mass. By reasoning similar to that adopted for estimating flight-hardware cost, we have used one-half the estimated mass of the full-scale system except for the nuclear systems. The nuclear system masses are assumed equal to the mass of the full-scale system. Launch-vehicle cost estimates were provided earlier in Table 6-2.

All cryogenic test flights are launched on a Delta. For these upper-stage tests, we reduce the Delta cost by $10 million to compensate for the absence of the existing upper stage, whose cost is included in Table 6-2.

6.2.7 Cost-Estimate Summary

Table 6-7 gives a summary of major estimated acquisition costs for each of the 15 detailed cases. The five nonrecurring cost estimates are summed to give a total of nonrecurring cost. The recurring cost (TFU) is based on a spacecraft sized to provide 2 kW electrical power to the satellite payload. This implies nonconstant payload masses. This summary does not include estimates of launch cost except for flight demonstration and does not consider the operational effectiveness of the various systems.
The DDT&E and TFU cost estimates were taken from the spreadsheets provided in Appendix F.

Table 6-6. Summary of the Estimated Cost for Flight Demonstration (FY95 $M)

Case No.	Demonstration Hardware Mass (kg/lb)	Launch Vehicle	Launch Cost	Demonstration Hardware Cost	Total Flight Demonstration Cost
1	1607/3536[a]	Taurus	19	15[c]	34
2	2321/5107[a]	Delta	56	25[c]	81
3	3220/7085[a]	Delta	56	24[c]	79
4	1268/2790[a]	Taurus	19	23[c]	41
5	1741/3830[a]	1/2 Delta	27	23[c]	50
6	3182/7001[a]	Delta	56	28[c]	83
7	3515/7733[a]	Delta	56	41[c]	97
8	8179/17,993[b]	Atlas IIAS	137	80[d]	217
9	11,348/ 24,965[b]	Titan IV	162	77[d]	239
10	5716/12,575[b]	Atlas 1	81	78[d]	159
11	6835/15,136[b]	Atlas IIA	99	76[d]	175
12	6841/15,051[b]	Atlas IIA	99	86[d]	185
13	10,654/ 23,438[a]	Delta	46[e]	39[c]	85
14	10,654/ 23,438[a]	Delta	46[e]	39[c]	85
15	10,654/ 23,438[a]	Delta	46[e]	39[c]	85

[a]50% wet mass of full-scale design minus payload
[b]100% wet mass of full-scale design minus payload
[c]50% TFU cost of full-scale design
[d]100% TFU cost of full-scale design
[e]Delta launch cost without transstage

6.2.8 Cost-Risk Assessment Methodology and Results

Cost-risk analysis comprises a series of engineering assessments and mathematical techniques, whose joint goal is to measure the degree of confidence in the "single best estimate" of system cost. A three-step procedure built upon results of a technical-risk study typically forms the cost-risk analysis. First, an engineering assessment of the various technologies involved in each subsystem leads to a triangular probability distribution of subsystem costs. Second, these subsystem cost distributions are sampled using Monte Carlo techniques to generate a cumulative distribution of total system cost. Finally, once the cumulative distribution has been established, the 50th, 70th, 90th or other cost percentiles can be read from the graph.

Table 6-7. Summary of the Estimated Acquisition Cost of the 15 Technology Combinations in the OECS Cost Spreadsheets (FY95 $M)

		Non-Recurring Costs					
Case No.	Technology Acquisition	Facilities & Equipment	DDT&E	Flight Demonstration	Shroud Modification	Estimated Total Non-recurring Cost	Estimated TFU Cost
1	29	25	91	34	0	179	30
2	29	25	180	81	0	315	50
3	29	25	119	79	0	252	47
4	29	25	118	41	0	213	45
5	25	25	122	50	0	222	46
6	45	60	211	83	0	399	55
7	16	60	278	97	0	451	82
8	35	225	975	217	20	1472	80
9	35	225	908	239	20	1427	77
10	35	225	913	159	20	1352	78
11	35	225	912	175	20	1367	76
12	37	180	857	185	40	1299	86
13	25	75	365	85	0	550	48
14	25	75	370	85	0	555	49
15	25	75	355	85	0	540	48

The cost-risk method employed in the OECS study establishes a credible and justifiable "risk factor" that is supported by engineering assessments of each subsystem, as well as other factors such as multiplicative error band of USCM 7 CERs, CER data scatter for various learning curves, alternate CERs, uncertainty in CER basis, alternate design solutions, different internal component environment, alternate manufacturing methods, unknown design details, technology readiness uncertainty, potential inheritance from similar technologies, potential applicability of other hardware, and uncertainty in NWODB approach. The risk factor is then applied to the best-estimate cost as a multiplicative factor to yield the low-end and high-end cost. Monte Carlo random sampling from each subsystem's triangular distribution produces a sequence of realizations of total-system cost that combine to define the cumulative distribution of total-system cost.

We have estimated the cost uncertainty for all development and TFU production CBS elements at three uncertainty levels: low, most likely, and high. The low cost estimate generally specifies subsystem cost under the most optimistic assumptions concerning development and production capabilities. The best-estimate cost is typically derived from the output of a cost model or other appropriate estimating procedure such as analogy or engineering buildup. The high-end cost encompasses the impacts of the many technical risks faced in developing and producing the subsystem. The resulting triangular cost distributions defined by the three parts are used as inputs to the Monte Carlo cost-risk analysis. An example of such a triangular distribution in presented in Figure 6-5.

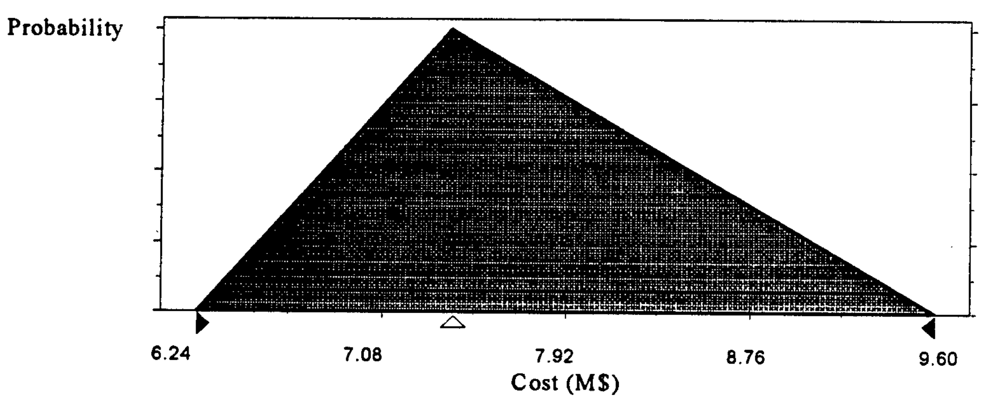


Figure 6-5. Typical triangular cost-distribution function.

Figure 6-6 shows the resulting cost distributions for development costs for Reference Case 1 based on 10,000 Monte Carlo samplings. Table 6-8 displays the 30%, 50%, and 70 % points for the development and production costs for all reference cases. The 50% confidence numbers do not agree with the nominal (most likely) values because most triangular distributions are skewed to the right due to risk considerations. In a triangular distribution skewed to the high side, the 50% confidence value (median) is always higher than the most likely (mode) value. Only in symmetrical distribution do the two values coincide. The cost-risk analysis for new technology concepts, with its bias to the higher cost side, produces conservative cost estimates.

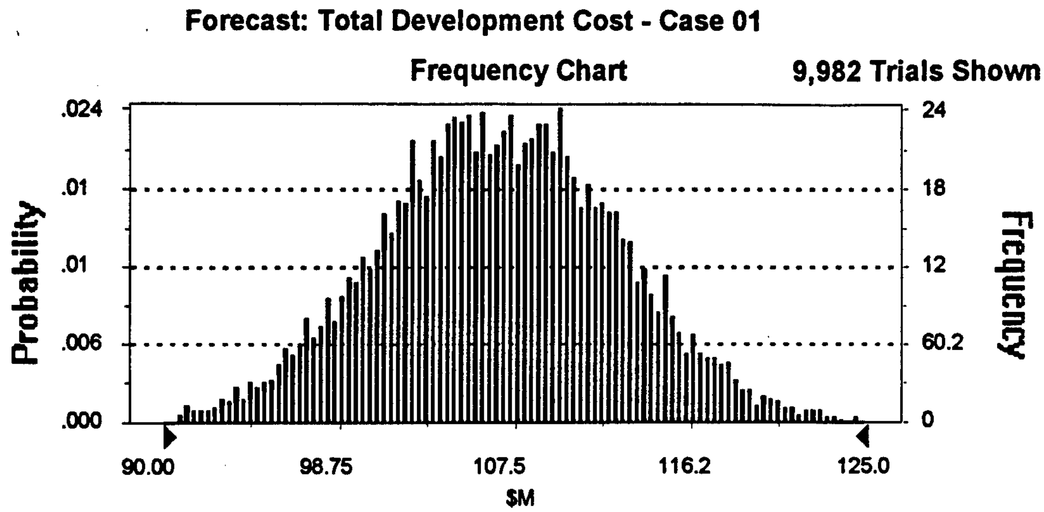


Figure 6-6. Example output for Case 1 of development cost.

Table 6-8. Summary of the Cost-Risk Estimates of the 15 Technology Combinations in the OECS Cost Spreadsheets (FY95 $M)

Spread-sheet No.	Percentiles of Production (TFU) Cost			Percentiles of Development Cost		
	30 %	50 %	70 %	30 %	50 %	70 %
1	33.92	35.24	36.50	103.60	106.75	109.90
2	50.6	52.88	55.47	204.19	212.61	220.04
3	47.24	49.48	51.83	139.58	143.77	147.98
4	47.92	50.14	52.34	134.36	138.46	142.64
5	50.55	52.96	55.21	139.83	144.08	148.56
6	57.82	59.82	61.83	227.31	236.42	245.13
7	88.19	91.75	95.54	318.71	330.37	342.07
8	84.97	89.55	94.91	964.94	1069.56	1180.06
9	83.40	88.01	93.40	894.85	993.78	1108.94
10	83.27	87.93	93.33	899.94	997.78	1108.91
11	83.33	88.06	93.81	1054.3	1184.9	1292.7
12	93.17	98.38	104.32	880.15	949.12	1019.35
13	51.79	53.99	56.32	418.42	440.40	464.19
14	52.31	54.46	56.79	425.57	447.94	471.54
15	51.92	54.09	56.38	408.99	430.84	453.24

The results for constant power level and variable payload mass allow us to draw the following qualitative conclusions:

- Development costs are low for solar electric and solar thermal concepts
- Development costs are medium for advanced cryogenic stages
- Development costs are high for nuclear reactor-based concepts
- Production costs are low for solar electric and solar thermal concepts
- Production costs are high for nuclear and solar bimodal concepts

7. COST/COST-EFFECTIVENESS RESULTS

Our cost-effectiveness analysis is a simple comparison of the cost of alternatives that perform equal tasks. In Section 4.3, we discuss the concept of constellation availability and its use to define equal constellation maintenance tasks independent of our technologies—specifically, independent of the deployment times of satellites to orbit. In this chapter we apply these concepts and define equal tasks based on buying and launching satellites to establish constellations and maintain them at a given availability for 15 years.

Many aspects of these constellations and their satellites are dependent on the ORM. In Chapter 2 we define the critical ORM parameters as:

- Satellite orbital parameters, which define lift ΔV
- Satellite reliability, which defines hold ΔV through mean mission duration (MMD)
- Satellite move ΔV
- Mass of payload (MPL)
- Electrical power to payload (PPL)
- Constellation size, which, with satellite reliability and other parameters, determines number of satellites and launchers bought

Using these parameters, we have defined representative satellite constellations for comparisons of the *relative* cost-effectiveness of the various technologies. Actual OECS cost estimates are shown in this chapter only for baseline technologies. Extensive parametric cost estimates for all ORMs and technologies appear in Appendix E. However, be cautioned that these numbers should also be used relative to one another and not as stand-alone estimates.

As discussed in Chapter 6, our cost is not a life cycle cost; it is an acquisition cost that excludes operations and support (O&S) costs associated with the satellite payload and bus. Our goal is to identify the circumstances for which each innovative technology either excels or is competitive with other technologies based on acquisition costs. This requires examining technologies over a range of ORM parameters.

We initially planned to rely on response-surface-methodology (RSM) analysis techniques to analyze and present the cost-effectiveness data. RSM can condense large discrete data sets into continuous approximation equations through multiple linear regression. Unfortunately, the launch cost component of the cost data is a step function, changing significantly with launch vehicle. Initial RSM approximations resulted in unacceptably large residuals when compared to the original data on which they were based. This problem defied attempts at solution, including varying the underlying experimental design of the RSM and reducing the number of independent parameters from five to three. The alternative of reducing the parameter ranges was not considered acceptable given the goals of the study, and it was impractical to subdivide the parameter ranges because of the corresponding increase in both input and output data.

Since RSM was not an option, we adopted the more usual but less flexible parametric analysis. Parametric analysis typically requires comparisons on a case-by-case basis or by simple curves of the dependent variable (cost in our case) as a function of a single independent variable (either MPL, PPL, MMD, move ΔV, or number of satellites required).

Our initial investigations led us to simplify our comparison process and consider only three independent variables: MPL, PPL, and number of satellites launched. We showed that MMD and move ΔV have significantly less influence on cost and cost differences than the other three variables. Thus, we have set MMD and move ΔV to nominal values while exploring the effects of changes in the MPL, PPL, and number of satellites launched.

For the cost estimates in Appendix E, we assign each of the three variables four values, dividing their ranges into three equal intervals. Thus, if the range of PPL is 0.5–5.0 kW, we examine discrete values of 0.5, 2.0, 3.5, and 5.0 kW. An exception is made for satellites required, where we have rounded to the nearest integer.

This format produces 64 combinations ($4 \times 4 \times 4$) of MPL, PPL, and number of satellites, providing a comprehensive picture of the relative cost of each technology for identical parameter values. However, these estimates are not usable for cost-effectiveness comparisons because the number of satellites bought for equal constellation maintenance tasks varies with satellite deployment time—i.e., with technology. This variation is considered in the cost-effectiveness tables of this chapter.

7.1 ORM 1 (GEO)

Table 7-1 shows the relative cost effectiveness of the baseline and innovative technologies for establishing and maintaining a constellation of five satellites in ORM 1 (GEO) for 15 years. Specific combinations of lift, hold, and move technologies are listed at the top of the table. These combinations were chosen as the least costly representatives for each category of lift technology based on the results in the cost tables in Appendix F. Table 7-1 also gives the number of satellites needed with each combination of technologies to establish and maintain the constellation. These numbers are averages of 500 replications of the GAP_PLUS simulation and are accurate to roughly ±0.2 satellites. Plus or minus 0.2 satellites corresponds roughly to ±1% of the total cost of establishing and maintaining the constellation—not a significant factor in the comparison. The numbers of satellites shown in Table 7-1 are approximately equal because all technologies require one on-orbit spare to achieve the 0.96 availability for which the table is constructed.

The left side of Table 7-1 shows four MPLs ranging from 200 to 2000 kg. The MPLs are arranged in four groups corresponding to four different values of PPLs. In terms of

Table 7-1. ORM 1 (GEO): Cost-Effectiveness Results for a Five-Satellite Constellation

			Technology											
	Baseline		Cryo	Nuc. Bimod.	Solar Bimod.	Solar Therm.	Nuclear Electric				Solar Electric			
Lift	Chem		Cryo	H_2 NB	H_2 SB	H_2 ST	NH_3 Arc	H_2 Arc	SPT	Xe Ion	NH_3 Arc	H_2 Arc	SPT	Xe Ion
Hold	N_2H_4 Arc		N_2H_4	N_2H_4	N_2H_4	N_2H_4Ar	NH_3 Arc	N_2H_4 Arc	SPT	Xe Ion	NH_3 Arc	N_2H_4 Arc	SPT	Xe Ion
Move	N_2H_4 Arc		N_2H_4	N_2H_4	N_2H_4	N_2H_4Ar	NH_3 Arc	N_2H_4 Arc	SPT	Xe Ion	NH_3 Arc	N_2H_4 Arc	SPT	Xe Ion
Req. # of Sat.	13.9		13.9	13.9	13.9	13.9	13.6	13.6	13.6	13.6	13.6	13.6	13.6	13.6
MPL (kg)	$B	LV[a]	PPL = 0.50 kW, MMD = 10 yr, Num Moves = 3.25											
200	1.82	D			L	L					L	L	L	L
800	3.41	A				D							D	22 L
1400	6.79	T		A		34 A						30 A	34 A	35 A
2000	7.49	T				28 A								28 A
			PPL = 2.0 kW, MMD = 10 yr, Num Moves = 3.25											
200	2.56	A	D		16 D	32 L					16 D	D	34 L	34 L
800	6.10	T		A	39 A	38 A				A	37 A	37 A	41 A	48 D
1400	6.88	T				32 A							32 A	33 A
2000	7.58	T												27 A
			PPL = 3.5 kW, MMD = 10 yr, Num Moves = 3.25											
200	2.64	A			D	D							17 D	32 L
800	6.18	T		A	38 A	37 A				A		35 A	39 A	47 D
1400	6.96	T				31 A							31 A	32 A
2000	7.66	T												26 A
			PPL = 5.0 kW, MMD = 10 yr, Num Moves = 3.25											
200	5.32	T	44 A	A	44 A	44 A			18 A	19 A	45 A	45 A	57 D	64 L
800	6.26	T			36 A	36 A				A			38 A	38 A
1400	7.03	T												30 A
2000	7.56	T												

[a]Launch vehicle: A = Atlas, D = Delta, L = LLV3, T = Titan. Launch vehicle indicated for innovative technologies only when step-down occurs.

- > 15% savings (% specified)
- within 15% of baseline cost
- > 15% additional cost
- technology too heavy for launch vehicle

standard moves (see Chapter 2), satellite MMD and maneuver ΔV are fixed at nominal values of 10 years and 3.25 moves, respectively.

The second column of Table 7-1 displays the OECS baseline acquisition costs in billions of FY95 dollars. All other entries in the table are relative to these costs. The column also identifies which baseline-technology launch vehicle (LV) would be required for each of the stipulated MPLs (Titan IV [T], Atlas [A], Delta [D], or LLV3 [L]).

The remainder of Table 7-1 is organized to highlight categories of relative costs. Four categories are considered:

- Greater than 15% savings relative to the baseline cost
- Within ±15% of the baseline cost
- More than 15% greater than the baseline cost
- No comparison possible, technology is too heavy for launch vehicle

Cost differences varying from the baseline by more than ±15% are considered significant. Smaller cost differences are considered indistinguishable from one another. This decision acknowledges the intrinsic uncertainty in estimating costs. While 15% is somewhat arbitrary, it can be justified by the fact that Table 7-1 (and the other tables of this chapter) are changed little if 10% or 20% is substituted for 15%.

The four categories are indicated in the table—and all other tables in this chapter—by different shading. Unshaded areas indicate combinations with greater than 15% savings over the baseline. Lightly shaded areas indicate combinations costing within ±15% of the baseline. Black indicates more than 15% greater cost than the baseline. Finally, the intermediate shading indicates the technology is too heavy for any launch vehicle. In addition to the shading, the letters A, D, L, and T identify the innovative technology launch vehicle whenever stepdown from the baseline launch vehicle occurs. Specific percentage savings are given when greater than 15%. Throughout this chapter, the reader can observe that greater than 15% savings are always accompanied by launch vehicle stepdown, although the converse is not true.

The results in Table 7-1 assume the RDT&E costs for each innovative technology are amortized over the number of satellites indicated. Thus, Table 7-1 indicates a very fast payback on technology investment. Opportunities for significant savings are almost exclusively found among the solar technologies, especially Xe ion, SPT, and solar thermal. Percentage savings over the baseline will increase if RDT&E costs can be amortized over more launches.

Table 7-2 is similar to Table 7-1 in every way except that a three- rather than a five-satellite constellation is represented. Because the amortization of the RDT&E costs are over a smaller number of satellites, there are fewer opportunities for significant savings, and savings for any given combination of technologies are less than in the five-satellite case.

Table 7-2. ORM 1 (GEO): Cost-Effectiveness Results for a Three-Satellite Constellation

	Baseline		Cryo	Nuc. Bimod.	Solar Bimod.	Solar Therm.	Nuclear Electric				Solar Electric			
Lift	Chem		Cryo	H_2 NB	H_2 SB	H_2 ST	NH_3 Arc	H_2 Arc	SPT	Xe Ion	NH_3 Arc	H_2 Arc	SPT	Xe Ion
Hold	N_2H_4 Arc		N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	NH_3 Arc	N_2H_4 Arc	SPT	Xe Ion	NH_3 Arc	N_2H_4 Arc	SPT	Xe Ion
Move	N_2H_4 Arc		N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	NH_3 Arc	N_2H_4 Arc	SPT	Xe Ion	NH_3 Arc	N_2H_4 Arc	SPT	Xe Ion
Req. # of Sat.	7.0		7.0	7.0	7.0	7.0	9.1	9.1	9.1	9.1	9.1	9.1	9.1	9.1
MPL (kg)	$B	LV[a]	PPL = 0.50 kW, MMD = 10 yr, Num Moves = 3.25											
200	1.00	D			L	L					L	L	L	L
800	1.84	A			A	D							D	L
1400	3.57	T		A		30 A						A	A	A
2000	3.96	T				24 A								A
			PPL = 2.0 kW, MMD = 10 yr, Num Moves = 3.25											
200	1.38	A	D		D	23 L					D	D	L	L
800	3.20	T		A	34 A	34 A				A	A	A	19 A	28 D
1400	3.63	T				28 A							A	A
2000	4.01	T												A
			PPL = 3.5 kW, MMD = 10 yr, Num Moves = 3.25											
200	1.43	A			D	D							D	L
800	3.25	T		A	33 A	32 A				A		A	17 A	26 D
1400	3.68	T				25 A							A	A
2000	4.06	T												A
			PPL = 5.0 kW, MMD = 10 yr, Num Moves = 3.25											
200	2.78	T	35 A	A	39 A	38 A			A	A	24 A	24 A	39 D	49 L
800	3.30	T			31 A	31 A				A			A	16 A
1400	3.72	T												A
2000	4.01	T												

[a]Launch vehicle: A = Atlas, D = Delta, L = LLV3, T = Titan. Launch vehicle indicated for innovative technologies only when step-down occurs.

Legend:
- > 15% savings (% specified)
- within 15% of baseline cost
- > 15% additional cost
- technology too heavy for launch vehicle

7.2 ORM 2A (GPS)

Table 7-3 shows the cost-effectiveness results for ORM 2a (GPS). The format of the table is identical to that of Tables 7-1 and 7-2, although the technology combinations and the values of the independent parameters differ. Table 7-3 is based on a constellation availability of 0.98. None of the innovative technologies require augmentation of the normal constellation with additional on-orbit spares to achieve this availability. This results in little variation in the number of satellites from technology to technology.

Many of the innovative technologies are capable of producing significant cost savings relative to the baseline for ORM 2a (GPS). Only the nuclear systems with their large RDT&E costs are left out entirely, while all the solar technologies are roughly equivalent in the savings they produce. The equivalency of the solar technologies can be explained by three factors: universal step-down from Delta or Atlas to LLV3; similar numbers of satellites required; and similar costs for technology RDT&E. Potential savings from using any of the solar technologies could be several billion dollars over 15 years from this one constellation.

7.3 ORM 3A (LEO-POLAR)

A table is not presented for ORM 3 (LEO-Polar). There are no occurrences of significant savings for this ORM and no instances of step-down. Lack of savings and step-down is attributable to the low ΔV values associated with this ORM. This does not automatically mean that the innovative technologies are not useful for lower altitude orbital transfers. It does suggest that low altitude applications are unlikely to offer an incentive for developing the technologies.

7.4 ORM 4 (HEO)

Table 7-4 presents the cost-effectiveness results for ORM 4 (HEO). This table corresponds to a two-satellite constellation with 0.97 availability. The format for this table is similar to that of the previous tables, but the technology choices differ for ORM 4. Neither nuclear electric nor solar electric technologies are suited to lift because of the high eccentricity of ORM 4. Therefore, they are not viable technology choices. Additionally, some of the combinations of payload mass and power exceed the lift capability of the baseline Titan IV (see Footnote b in the column on baseline cost). There is no baseline cost estimate for these cases and consequently no way of comparing relative costs. These rows are left unshaded for the innovative technologies that can lift the satellite to HEO. However, only their launch vehicle is indicated.

Because the HEO constellation contains only two satellites, the number of satellites required is small. This, combined with a lift ΔV smaller than ORM 1 (GEO), results in few instances of significant cost savings.

Table 7-3. ORM 2a (GPS): Cost-Effectiveness Results

	Technology													
	Baseline		Cryo	Nuc. Bimod.	Solar Bimod.	Solar Therm.	Nuclear Electric				Solar Electric			
Lift	Chem		Cryo	H_2 NB	H_2 SB	H_2 ST	NH_3 Arc	H_2 Arc	SPT	Xe Ion	NH_3 Arc	H_2 Arc	SPT	Xe Ion
Hold	Chem		Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	N_2H_4 Arc	Chem	Chem
Move	Chem		Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	N_2H_4 Arc	Chem	Chem
Req. # of Sat.	57.7		57.7	57.7	57.5	57.5	56.8	56.8	56.8	56.8	56.8	56.8	56.8	56.8
MPL (kg)	$B	LV[a]												
PPL = .5 kW, MMD = 10 yr, Num Moves = 3														
200	5.91	D			26 L	28 L					29 L	27 L	29 L	28 L
300	6.29	D			24 L	25 L					26 L	24 L	27 L	26 L
400	6.68	D			23 L	24 L					25 L	23 L	25 L	24 L
500	9.66	A	18 D		43 L	44 L					44 L	43 L	44 L	43 L
PPL = .83 kW, MMD = 10 yr, Num Moves = 3														
200	5.96	D			26 L	27 L					28 L	26 L	29 L	28 L
300	6.34	D			24 L	24 L					26 L	24 L	26 L	26 L
400	6.73	D			23 L	23 L					24 L	23 L	24 L	23 L
500	9.71	A	18 D		43 L	43 L					44 L	43 L	44 L	43 L
PPL = 1.17 kW, MMD = 10 yr, Num Moves = 3														
200	6.02	D			26 L	26 L					28 L	26 L	29 L	28 L
300	6.44	D			24 L	24 L					26 L	24 L	26 L	25 L
400	9.44	A	19 D		45 L	44 L					45 L	44 L	46 L	45 L
500	9.77	A	18 D		43 L	43 L					43 L	42 L	44 L	43 L
PPL =1.50 kW, MMD = 10 yr, Num Moves = 3														
200	6.11	D			26 L	26 L					28 L	26 L	28 L	28 L
300	9.14	D	19 D		46 L	46 L					47 L	46 L	47 L	47 L
400	9.49	A	18 D		44 L	44 L					45 L	44 L	45 L	44 L
500	9.82	A	18 D		43 L	42 L					43 L	42 L	43 L	42 L

[a]Launch vehicle: A = Atlas, D = Delta, L = LLV3, T = Titan. Launch vehicle indicated for innovative technologies only when step-down occurs.

(white)	> 15% savings (% specified)
(light shading)	within 15% of baseline cost
(black)	> 15% additional cost
(dark shading)	technology too heavy for launch vehicle

Table 7-4. ORM 4 (HEO): Cost-Effectiveness Results

	Technology										
	Baseline		Direct	Cryo		Nuclear Bimodal		Solar Bimodal		Solar Thermal	
Lift	Direct		Direct	Cryo	Cryo	H_2 NB	H_2 NB	H_2 SB	H_2 SB	H_2 ST	H_2 ST
Hold	Chem		N_2H_4 Arc	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem
Move	Chem		N_2H_4 Arc	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem
Req. # Sat.	5.3		5.3	5.3	5.3	5.3	5.3	5.4	5.4	5.4	5.4
MPL (kg)	\$B	LV[a]	PPL = 0.5 kW, MMD = 7.5 yr, Num Moves = 2								
500	1.53	A		D	D			D	D	L	L
1000	2.02	A		D	D			D		D	D
1500	3.10	T		A	A			A	A	A	A
2000	3.50	T									
			PPL = 2.0 kW, MMD = 7.5 yr, Num Moves = 2								
500	1.58	A		D	D			D	D	D	D
1000	2.72	T	24 A	A	A			A	A	A	A
1500	3.15	T						A	A	A	A
2000	b					T	T	T	T	A	A
			PPL = 3.5 kW, MMD = 7.5 yr, Num Moves = 2								
500	1.62	A		D				D	D	D	D
1000	2.76	T		A				A	A	A	A
1500	3.19	T						A	A	A	A
2000	b					T	T	T	T	A	A
			PPL =5.0 kW, MMD = 7.5 yr, Num Moves = 2								
500	2.31	T	28 A	A	A			17 A	17 A	17 A	17 A
1000	2.80	T						A	A	A	A
1500	b					T	T	A	T	A	A
2000	b					T	T	T	T	T	T

[a]Launch vehicle: A = Atlas, D = Delta, L = LLV3, T = Titan. Launch vehicle indicated for innovative technologies only when step-down occurs.
[b]The baseline technology was too heavy for Titan IV; therefore, the percentage of savings over the baseline cannot be given for the innovative technologies.

- > 15% savings (% specified)
- within 15% of baseline cost
- > 15% additional cost
- technology too heavy for launch vehicle

7.5 COST-EFFECTIVENESS SUMMARY

The cost-effectiveness results are easily summarized:

- For equal tasks, solar lift technologies are frequently less costly than the baseline technologies despite longer satellite deployment times. This cost reduction is always associated with launch vehicle step-down.
- Development and use of one or more innovative solar technologies for ORM 1 (GEO) and/or ORM 2a (GPS) lift is justifiable based on cost effectiveness.
- Use of an innovative technology developed for GEO and/or GPS is likely to be cost effective for ORM 4 (HEO) lift.
- Use of an innovative technology developed for GEO and/or GPS may or may not offer significant cost advantages when used at LEO.

The first two conclusions are unequivocal within the context of the OECS. The real world is more complex. For example, payloads that might ideally go on a Titan IV if cost were not a consideration may be pared down to fit on an Atlas IIAS. The performance and dollar costs associated with such compromises are hard to quantify in either operational or cost terms. Does this mean that the above conclusions don't hold in the real world? No. There is little doubt that the current GPS function could be launched on LLV3, and many payload functions going to GEO on Atlas or Titan could be stepped down.

8. SUMMARY AND CONCLUSIONS

Chemical upper stages are our only current means for moving satellites from booster burnout to high altitude orbits. However studies indicate a number of innovative upper-stage propulsion technologies may achieve better performance than chemical propulsion technologies on a given booster. Innovative technologies also offer opportunities to move payloads to smaller and less expensive boosters, albeit often with longer orbital transfer times. The development of these technologies moved slowly in an environment in which operational costs were not a primary consideration. That situation has changed—reducing operational cost is now a priority, and interest in the innovative technologies is growing.

Given this new cost consciousness, it is sensible to compare the innovative technologies to today's baseline. The *Space Propulsion and Power: Operational Effectiveness and Cost Study (OECS)* does that for six innovative space-propulsion technologies using a consistent methodology to examine effectiveness and cost effectiveness. The baseline for comparison consists of appropriate configurations of the Delta, Atlas, and Titan IV with Solid Rocket Motor Upgrade.

The OECS focuses on upper stage propulsion (lift), but it also considers propulsion for on-orbit stationkeeping (hold) and maneuver (move). Electrical power generation is an integral part of the comparison because many of the innovative lift technologies provide electrical power with propulsion. Each of the innovative technologies considered here is estimated to be capable of supporting flight demonstration within seven years and initial operational capability (IOC) within ten years if the technology is adequately funded.

The innovative technologies considered in this study are:

- Advanced cryo (advanced cryogenic propulsion and photovoltaic power)
- Nuclear bimodal (nuclear thermal propulsion and thermoelectric power)
- Solar bimodal (solar thermal propulsion and thermionic power)
- Solar thermal (solar thermal propulsion and photovoltaic power)
- Nuclear electric (nuclear electric propulsion and thermionic power)
- Solar electric (solar electric propulsion and photovoltaic power)

Because of the diversity of electric propulsion thrusters, four thruster options for lift are considered for nuclear and solar electric: ammonia arcjets, hydrogen arcjets, xenon stationary plasma thrusters, and xenon ion thrusters.

The principal advantage of the innovative technologies over the baseline technologies is their higher I_{sp} values. These values translate to less propellant mass to produce a given change in velocity—that is, more efficient use of propellant. The principal disadvantages of the innovative technologies are:

- Large propulsion-system mass (characteristic of nuclear bimodal and nuclear electric)

- Low thrust, which results in long transfer times (solar bimodal and solar thermal transfer to GEO ≈ 1 mo, nuclear and solar electric transfer to GEO ≈ 10 mo)
- Reliance on cryogenic hydrogen as a propellant, which results in large propellant tanks and, hence, large propulsion systems that may not always fit within existing launch fairings (nuclear bimodal, solar bimodal, solar thermal, and nuclear and solar electric hydrogen arcjets)

This study uses four representative operational reference missions (ORMs) as a framework for the effectiveness and cost-effectiveness comparisons:

- ORM 1: geosynchronous Earth orbit (GEO), as typified by DSCS
- ORM 2: mid-Earth orbit (MEO), as typified by GPS
- ORM 3: low Earth orbit (LEO), as typified by DMSP
- ORM 4: highly eccentric orbit (HEO), as typified by Molniya

These ORMs are based primarily on current and potentially useful military constellations.

The effectiveness analysis focuses on two factors: (1) the ability of all the technologies to place payload mass and payload electrical power in mission orbit, and (2) the ability of the innovative technologies to step down payloads from their baseline launch vehicles to less capable, less expensive launch vehicles. A single computer design model was used to incorporate all the innovative propulsion and power technologies into the designs of the upper stages and satellite buses.

The cost model combines the CERs developed for the innovative technologies with the CERs for the existing upper stage and satellite components. This model estimates system acquisition costs rather than life cycle costs. The acquisition costs differ from life cycle costs by the omission of operation and support (O&S) costs for the satellite payload and bus, both of which should be essentially independent of the technologies being considered. All other costs associated with life cycle costs are considered, including technology RDT&E, launch vehicle acquisition and launch support, and satellite bus and payload acquisition. The analysis compares the cost-effectiveness of the technologies in establishing and maintaining equivalent constellations for 15 years in each ORM.

8.1 EFFECTIVENESS RESULTS

The results of the effectiveness analysis show the following:

- Innovative technologies can provide substantial increases in the on-orbit payload mass and electrical power of satellites in high energy ORMs (GEO, MEO, and HEO). Thus the innovative technologies could eliminate any need for a highly optimized booster.
- Launch packaging for current Atlas and Delta fairings is a potentially serious problem for all the low thrust technologies.

- No innovative technology can enable complete payload step-down from a fully loaded GEO-bound Titan IV to an Atlas. However a booster capable of about 50–60% of the Titan IV performance to LEO can support complete step-down for the payloads on all Titan IV-class launch vehicles.
- For most other high energy ORMs, one or more of the innovative technologies would permit payloads that exceed current capacities while stepping down from Titan IV to Atlas, from Atlas to Delta, or from Delta to Martin-Lockheed Launch Vehicle 3.
- High constellation availabilities can be maintained with very low thrust electric technologies despite the long trip times required, provided on-orbit spare satellites are used.

8.2 COST-EFFECTIVENESS RESULTS

We assume acquisition costs for the innovative technologies are significantly different from baseline costs if they differ by more than 15%. The cost-effectiveness results show the following:

- Solar lift technologies are frequently less costly than baseline technologies for equal tasks, despite longer satellite-deployment times arising from the lower thrust.
- Launch vehicle step-down is the essential element in realizing cost savings from the innovative technologies.
- Developing and using one or more innovative solar technologies for ORM I (GEO) and ORM2a (GPS) lift would be cost effective.
- No innovative technology provides significant cost savings for the LEO ORM.

8.3 GENERAL TECHNOLOGY ASSESSMENTS

The following general technology assessments are based on the results of the effectiveness and the cost-effectiveness analyses:

- *Solar thermal* and *solar bimodal systems* are the most pervasively cost-effective innovative technologies; they have short trip times; and they perform very well.
- *Solar electric systems* are very competitive for all ORMs except HEO, unless the long trip times they require prohibit their use for a particular mission.
- *Nuclear bimodal* and *nuclear electric systems* perform effectively on Titan and Atlas, but their high mass prevents them from performing effectively on Delta. However these technologies are not typically cost effective because of their large development costs.
- *Advanced cryogenic systems* provide only incremental improvements in performance except for Delta, where dramatic results could be achieved by replacing the second stage and the PAM II upper stage with a cryogenic stage.

8.4 FINAL THOUGHTS

The results of the OECS could—and probably should—have a substantial impact on any future programs to develop launch vehicles. In particular, the innovative technologies may impose constraints on ground processing facilities, on the launch requirements to LEO orbits, and on fairing dimensions.

Will the development of the evolved expendable launch vehicle (EELV) alter the conclusions of the OECS? There will be less savings from adopting an innovative technology if the EELV provides a significant reduction in launch costs, especially if it reduces the difference in cost between adjacent classes of launch vehicles. As it stands today, there is very roughly a difference of a factor of two in the cost of adjacent classes of launch vehicle. For example, there is over $100M difference in cost between Atlas IIAS and Titan IV. If the EELV reduced this difference to $10M instead of $100M, it is doubtful whether the improved cost effectiveness of the innovative technologies would be a sufficient incentive to develop them. However the innovative technologies would still offer improved lift capability over chemical upper stages, and this alone might justify their development and use.

APPENDIX A

RELATIONSHIP BETWEEN MEAN MISSION DURATION AND DESIGN LIFE IN THE OECS

Satellite reliability is generally modeled with the two-parameter Weibull distribution. Weibull reliability is given by

$$R(t) = e^{-\left(\frac{t}{\alpha}\right)^{\beta}}$$

where t is time, the parameter α is related to the mean, and β is the "shape" parameter. However, the OECS assumes that R(t) goes to 0 at the mean satellite design life (i.e., at truncation time), τ. By fixing satellite reliability at design life at 0.6, we get a fixed relationship between satellite mean mission duration (MMD) and design life. This reduces the number of analysis parameters we must consider in the OECS without any significant loss of generality.

Assuming $\beta = 1.6$ (a typical value for general analyses) and $R(\tau) = 0.6$, we have

$$0.6 = e^{-\left(\frac{\tau}{\alpha}\right)^{1.6}}$$

By taking logs of both sides and rearranging, we get

$$(-\ln 0.6)^{1/1.6} = 0.6572 = \frac{\tau}{\alpha}$$

whence

$$\tau = 0.6572\alpha$$

or

$$\alpha = 1.5217\tau$$

Now the mean mission duration, MMD is given by

$$MMD(\alpha,\tau) = \int_0^{\tau} e^{-\left(\frac{t}{\alpha}\right)^{\beta}} dt$$

or substituting for α

$$MMD(\tau) = \int_0^\tau e^{-\left(\frac{0.6572t}{\tau}\right)^{1.6}} dt$$

We now ask the question, what value of τ satisfies the equation

$$f(\tau) = MMD - \int_0^\tau e^{-\left(\frac{0.6572t}{\tau}\right)^{1.6}} dt = 0$$

We solve for τ by writing k = 0.6572 and transforming the integral with the substitution

$$x = \frac{kt}{\tau}$$

whence

$$\frac{\tau}{k}\int_0^k e^{-x^{1.6}} dx = 0.8311\tau$$

This yields a linear equation in τ

$$\tau = \frac{MMD}{0.8311}$$

and

$$\alpha = 1.5217\tau = 1.8309 \times MMD$$

APPENDIX B

SIZING RELATIONSHIPS IN THE COST/ENGINEERING MODEL (OCEM)

OCEM is a complex engineering and costing model. It performs a high-level design of (that is, it sizes) a complete spacecraft having lift, hold, move, and electrical power subsystems designed to perform a specified mission. In one mode, OCEM maximizes a performance parameter, such as satellite payload mass. In a second mode, it minimizes cost by determining the smallest possible launch vehicle that will support the mission. Both objectives involve many sizing iterations as the model converges on a solution.

This appendix focuses on the OCEM sizing process. It overviews methodology and assumptions and how they are applied to each of the OECS technologies. It also discusses the sizing algorithms used by the model.

B.1 OVERVIEW OF OCEM SIZING

The sizing portion of OCEM is based on the spacecraft sizing and launch vehicle sizing algorithms developed by the Aerospace Corporation's Vehicle Design and Manufacturing Department and on additional innovative technology-specific algorithms provided by the OECS technologists. The model performs a preliminary sizing of the satellite bus and any integrated upper stage given satellite mission and payload requirements. These requirements are

- Mission orbit (that is, the ORM)
- Lifetime
- On-orbit maneuvers
- Payload mass
- Payload end-of-life electric power
- Pointing accuracy

By payload we refer to all nonbus hardware required to perform an operational mission (see Section 1.5 of Chapter 1). Examples of payload hardware include mission sensors, antennas, amplifiers, lenses, and focal plane coolers. The bus includes structure, command and control, power, thermal control, propulsion, and other housekeeping subsystems.

Most of the technologies are scalable, that is, the design can be customized for a specific set of requirements. However baseline chemical, advanced cryo, and nuclear bimodal lift are fixed designs. In addition, several technologies use an integral propulsion subsystem for both orbit transfer and on-orbit propulsion. Exceptions are baseline chemical, advanced cryo, and solar thermal. Baseline and advanced cryo use a separate upper stage. When orbit transfer is complete, the upper stage is abandoned. Solar thermal involves another approach: its transfer propulsion is integrated to the spacecraft (i.e., it uses spacecraft resources such as power and command and control), but the subsystem is separated from the satellite once transfer is complete. The same is true for solar electric-hydrogen arcjet lift because of the large size of the hydrogen tank. Table B-1 summarizes these approaches.

Table B-1. Sizing Approaches

Technology	Design Type	How Used
Baseline Chemical	Fixed	Separate upper stage; one per launch vehicle
Advanced Cryo	Fixed	Separate upper stage; one per launch vehicle
Nuclear Bimodal	Fixed	Integral to spacecraft
Solar Bimodal	Scalable	Integral to spacecraft
Solar Thermal	Scalable	Integrated with spacecraft (drops off)
Nuclear Electric	Scalable	Integral to spacecraft
Solar Electric	Scalable	Integral to spacecraft (H_2 arcjet lift propulsion drops off)

Mission and payload requirements are used to size eight major spacecraft subsystems: payload, attitude determination and control (ADACS), telemetry tracking and commanding (TT&C), command and data handling (C&DH), thermal, structure, power, and propulsion. The propulsion subsystem provides transfer and on-orbit propulsion. Table B-2 lists the general sizing methodology and major assumptions used for each of the major subsystems. The overall process of sizing the spacecraft is summarized in Figure 4-6 of Chapter 4. The process involves several iterations to converge on a design because all the subsystems are interrelated.

Table B-2. Sizing Method and Assumptions for Spacecraft Subsystems

Subsystem	Sizing Method	Major Assumptions
Payload	Fixed mass and power (as input)	User input
ADACS	Database lookup	0.07° attitude knowledge 0.01° pointing accuracy
TT&C	Database lookup	Standard Ground Link System (SGLS)-based downlink
C&DH	Database lookup	Integrate spacecraft processor (also performs ADACS processing) No data storage Mil-Std-1553B data bus
Thermal	Historical spacecraft properties	Passive thermal control system (radiators, heat pipes, etc.)
Structure	Historical spacecraft properties	Aluminum with selective use of composites
Power	Analytical relationships	Technology-specific
Propulsion	Analytical relationships	Technology-specific
Spacecraft Bus	Sum of propulsion, ADACS, TT&C, C&DH, thermal, power, and structure	15% growth margin based on dry spacecraft bus
Launch Vehicle Adapter	Fixed mass	68 kg

B.2 COMMON SIZING ASSUMPTIONS FOR SATELLITE SUBSYSTEMS

Many common assumptions about sizing can be made that are not dependent upon the propulsion and power subsystems. These assumptions are discussed in this section.

B.2.1 Payload

The payload is treated as a black box by OCEM; only payload mass and payload electric power need to be specified. These parameters are input by the user in the case of the cost analysis or optimized by OCEM when determining effectiveness (see Figure 4.1 in Chapter 4). Payload mass and power influence the rest of the spacecraft design.

B.2.2 Attitude Determination and Control System (ADACS)

ADACS is sized using a list of currently available components chosen by Aerospace Corporation. The subsystem, which is identical for all satellites, consists of four fine digital sun sensors (2-axis), two conical Earth-scanning sensors, two gyros, two reaction wheel assemblies, and interface electronics. LEO satellites require the addition of a torque rod and magnetometer, and HEO satellites require the addition of three rate-measuring assemblies to monitor changes in pitch rates.

B.2.3 Telemetry, Tracking, and Commanding (TT&C)

The TT&C subsystem is sized using a list of currently available components chosen by Aerospace Corporation. All satellites contain identical components. It is SGLS-based and consists of three antennas, two transponders, two radio frequency (RF) diplexers, and two command/telemetry units.

B.2.4 Command and Data Handling (C&DH)

C&DH is sized using a list of currently available components chosen by Aerospace Corporation. All satellites contain identical components. It consists of two integrated computers (shared with the ADACS) and a Mil-Std-1553 data bus.

B.2.5 Thermal

The thermal subsystem is used to dissipate heat generated by the power subsystem and is thus sized based on end-of-life (EOL) power. Both subsystem mass and power requirements are sized as a percentage of EOL power based on historical satellite systems.

B.2.6 Structure

Structure is sized as a percentage of the dry satellite mass and is based on historical data. Dry mass refers to the bus structure in our case. Some of the innovative propulsion and power technologies incorporate their own structure, and this mass is not included. Solar electrics use the entire dry spacecraft excluding the transfer hydrogen arcjets and the hydrogen tank, which are dropped off. For nuclear electrics, the nuclear system is excluded from the satellite mass, as is the hydrogen arcjet propellant tank. The hydrogen tank is also excluded for the solar thermal, solar bimodal, and nuclear bimodal technologies, as are the transfer propulsion systems. The solar thermal propulsion system and its propellant tank drop off, and solar bimodal and nuclear bimodal incorporate their own structure. The separate upper stages used with the baseline chemical and advanced cryogenic systems are not part of the on-orbit satellite. These stages have their own structure.

B.2.7 Bus Margin

A 15% margin is applied to all spacecraft bus masses to account for uncertainties in the technologies and for possible growth. Spacecraft historically undergo a 25% increase in their mass from concept to production. The 15% spacecraft margin is based on the rationale there is no payload mass uncertainty in the OECS; thus, less bus mass uncertainty is reasonable. A 15% margin is also applied to the advanced cryogenic upper stages. Sandia Labs' experience with launch vehicles and the National Launch System (NLS) program indicates a growth rate of 10%, although the rate may be somewhat higher.

B.2.8 Propellant

Propellant sizing is based on orbit transfer and on-orbit stationkeeping and maneuver requirements. To minimize spacecraft mass, tanks are shared whenever possible. OCEM uses the rocket equation. Transfer propellant is sized from the mass required to be transferred to final orbit, as appropriate for each technology and mission scenario. On-orbit propellant is sized from spacecraft on-orbit dry mass (e.g., after a separate upper stage has been dropped off).

Residuals are added to the transfer propellant. Reflecting current practices, bipropellant transfers have a 5% residual based on the transfer ΔV requirement. For electric propulsion transfers, there is a 10% propellant residual for xenon (based on propellant mass), 6% for ammonia, and 7% for hydrogen. The hydrogen for solar thermal, solar bimodal, and nuclear bimodal also has a 7% residual. The residual for the advanced cryogenic upper stage is calculated from an Aerospace Corporation weight-estimating relationship (the residual is about 1%).

Propellant tank volumes are also increased to account for *ullage*. Ullage is the tank volume that cannot be filled because of filler location, trapped air in the tank, etc. Ullage percentages are 3% for hydrogen, 12% for xenon, and 10% for all other propellants.

B.2.9 Launch Vehicle Adapter

The launch vehicle adapter is assumed to be 68 kg for all spacecraft and all launch vehicles. This is a reasonable simplifying assumption.

B.3 BASELINE CHEMICAL/DIRECT SYSTEMS

The baseline chemical systems are comprised of two parts: an upper stage (which is not required by all ORMs) and the spacecraft.

B.3.1 Upper Stages

The upper stage is not sized by OCEM—existing stages are used. The need for an upper stage is determined by the ORM and launch vehicle (Table B-3). The spacecraft has a small, integrated bipropellant-propulsion subsystem to complete the orbital transfer in cases where the launch vehicle does not place the spacecraft into its required orbit (that is, the launch vehicle only goes to GEO transfer orbit [GTO]).

Table B-3. Current Upper Stages Used By OCEM

ORM	Titan IV	Atlas IIAS	Delta II
GEO	Centaur upper stage to GEO	Centaur to GTO[b]	PAM-D upper stage to GTO
MEO	N/A	Centaur to MEO TO[c]	PAM-D to MEO TO
LEO	NUS[a]	Centaur to LEO	Delta direct to LEO/ LEO TO[d]
HEO	NUS to orbit slightly lower than HEO	Centaur to HEO	PAM-D upper stage to HEO

[a]NUS = no upper stage
[b]GTO = GEO transfer orbit
[c]MEO TO = MEO transfer orbit
[d]LEO TO = LEO transfer orbit

B.3.2 Spacecraft Sizing

The spacecraft used with the baseline systems are very conventional. The general subsystems are sized as described above. The power subsystem uses advanced rigid gallium arsenide (GaAs) arrays, and propulsion is provided by monopropellant, bipropellant, or hydrazine (N_2H_4) arcjets.

B.3.2.1 POWER SUBSYSTEM

Power is provided by advanced, rigid, multi-junction GaAs arrays with a 9.1-kg deployment mechanism. Array characteristics include 21% efficiency and 245.84 W/m^2. For HEO, 30-mil frontal cover glass is assumed (34.87 W/kg); all other ORMs assume 4-mil cover glass (47.59 W/kg). Energy storage, sized for eclipsing, is provided by nickel hydride (NiH_2) common pressure vessel (CPV) batteries (49 W-hr/kg). The remaining power component is the power management and distribution system (PMAD), consisting of regulators/converters and wiring harnesses. It is sized from historical satellite systems analyzed by Aerospace Corporation.

The solar cells are sized based on satellite beginning-of-life (BOL) power requirements. BOL power is EOL power divided by solar cell degradation, which is determined from the ORM and the satellite's required (design) life.

B.3.2.2 PROPULSION SUBSYSTEM

An integral bipropellant subsystem provides on-orbit transfer for those missions requiring an additional boost. On-orbit propulsion options are summarized in Table B-4, and the subsystems are discussed below. Tanks are shared as much as possible to reduce spacecraft mass. The reaction control system (RCS) plumbing is sized as a percentage of dry spacecraft mass based on historical data.

Table B-4. On-Orbit Thruster Options Considered with Chemical Lift

ORM	Stationkeeping	On-orbit maneuver
GEO	8 monoprop N_2H_4 for E-W and 4 N_2H_4 arcjets for N-S	N_2H_4 arcjet *or* Biprop $N_2H_4 + N_2O_4$
MEO	12 monoprop N_2H_4	Monoprop N_2H_4
LEO	12 monoprop N_2H_4 *or* 4 N_2H_4 arcjets and 8 monoprop N_2H_4 *or* 4 biprop $N_2H_4 + N_2O_4$ and 8 monoprop N_2H_4	Same as stationkeeping
HEO	4 N_2H_4 arcjet and 8 monoprop N_2H_4 *or* 12 monoprop N_2H_4 *or* 12 biprop MMH + N_2O_4	Same as stationkeeping

Bipropellant Chemical

The bipropellant propulsion subsystem uses N_2H_4 and N_2O_4 if the spacecraft has N_2H_4 already on board; it uses MMH and N_2O_4 otherwise. Propellant tanks are aluminum and are sized as a percentage of propellant mass based on historical data. The thrusters are off the shelf. The transfer thrusters have an I_{sp} of 311 s; those used for stationkeeping have an I_{sp} of 289 s. A helium pressurant is used with the system. These tanks are composite and are sized for the system.

Monopropellant Chemical

The monopropellant N_2H_4 used for minor stationkeeping and reaction control is a small off-the-shelf thruster. The thruster has an I_{sp} of 225 s. The N_2H_4 tank and associated plumbing comprise the rest of the subsystem.

Hydrazine Arcjets

The N_2H_4 arcjets are also based on an off-the-shelf system. The thrusters are 1800 W and have an I_{sp} of 500 s. Power conditioning is based on the specific thruster requirements. The tank and plumbing comprise the rest of the subsystem. The arcjets are run with off-duty cycle power so they do not add to the spacecraft power requirements, a common practice.

B.4 ADVANCED CRYOGENIC

Like the baseline chemical systems, the advanced cryo systems consist of two parts: an upper stage (as required by the ORM and launch vehicle) and the on-orbit spacecraft.

B.4.1 Advanced Cryo Upper Stage

The orbits and launch vehicles using an advanced cryogenic stage are summarized in Table B-5. Three stages were developed—one for each launch vehicle. The stages were designed with the following in mind:

- For Titan IV, the goal was to have a stage roughly the same height as the current Centaur (9.0 m). The new upper stage is actually one meter taller due to the size of the engine.
- For Atlas IIAS, the new stage replaces the Centaur and was sized to maximize the Atlas' performance to GTO.
- For Delta II, the new stage replaces the existing Delta second stage and the PAM upper stage with a total propellant mass of 15,910 kg (35,000 lb). Such a system is similar to McDonnell Douglas' Delta III concept or the Delta growth option mentioned in the 1991 edition of the AIAA *International Reference Guide to Space Launch Systems* (Isakowitz, p. 216).

Table B-5. Use of Advanced Cryo Upper Stages for Orbital Transfer

ORM	Titan IV	Atlas	Delta
GEO	Titan adv cryo upper stage to GEO	Atlas adv cryo stage to GTO	Delta adv cryo stage to GTO
MEO	N/A	Atlas adv cryo stage to MEO TO	Delta with adv cryo stage to MEO TO
LEO	NUS	Atlas adv cryo stage direct to LEO or LEO TO	Delta adv cryo stage to LEO or LEO TO
HEO	NUS to orbit slightly lower than HEO	Atlas adv cryo stage to HEO	Delta adv cryo stage to HEO

A summary of each stage is found in Table B-6. The propellant mass fractions of each stage are certainly reasonable. As a comparison, the Atlas Centaur value is 0.885. That of the Titan Centaur stage is 0.852. (This stage is considered by most sources to be heavier than necessary for Titan application, since it was originally designed for use in the Space Shuttle). Each stage has the components described below.

Table B-6. Design and Performance Summary of OECS Advanced Cryo Designs

	Titan IV	Atlas IIAS	Delta II
Masses:			
Dry Mass	2013 kg (4429 lb)	2200 kg (4841 lb)	2003 kg (4408 lb)
Propellant (total)	20,677 kg (45,490 lb)	23,770 kg (52,293 lb)	16,089 kg (35,396 lb)
Propellant (usable)	19,841 kg (43,650 lb)	23,507 kg (51,716 lb)	15,9090 kg (35,000 lb)
Total Stage Mass	22,691 kg (49,920 lb)	25,970 kg (57,134 lb)	18,093 kg (39,804 lb)
Propellant Mass Fraction	0.874	0.905	0.879
Performance to:			
GEO or GTO	6730 kg (14,807 lb) GEO	4679 kg (10,293 lb) GEO TO	3330 kg (7326 lb) GEO TO
GPS or GPS TO	N/A to GPS	5001 kg (11,003 lb) GPS TO	3604 kg (7928 lb) GPS TO
LEO	N/A to LEO	9621 kg (21,167 lb)	7806 kg (17,174 lb)

B.4.1.1 PROPULSION SUBSYSTEM

The propulsion subsystem is comprised of an engine and plumbing. The engine is a Rocketdyne 45,000 lb_f IME, which contains nozzles, thrust chambers, injectors, pumps, etc. All three upper stages use the same engine. The plumbing includes feed lines around engines to tank, valving, etc. They are sized using Aerospace-developed weight estimating relationships (WERs).

The engine is an advanced cryogenic engine studied by Rocketdyne: it has not been built. Table B-7 gives a comparison of this engine to the RL-10A-4 used in the Atlas Centaur stage (the Titan Centaur stage uses the RL-10A-3) and the Russian D-57. The IME is a reasonable engine; if it had a thrust-to-weight ratio of 50.6:1, the engine would be 64 lb heavier. Given that we are comparing the IME to a Russian engine and to an engine based on 20-year-old technology, a 64-lb reduction should not be difficult. Molded components (such as turbopumps) and other advances can reduce the mass.

B.4.1.2 STRUCTURE

The structure is comprised of tanks and the thrust structure (i.e., what holds the engine to the stage). The Atlas and Delta stages also have an intertank structure; the Titan stage uses common bulkhead tanks (to keep its height similar to the current Centaur). All items are sized using Aerospace-developed WERs.

B.4.1.3 THERMAL CONTROL

Thermal control includes blankets, MLI, and foam. The Titan version contains more insulation for boil-off control since it is used to transfer from GTO to GEO. The subsystem is sized using Aerospace-developed WERs.

Table B-7. Comparison of Proposed Rocketdyne Cryo Stage Engine to Existing Cryo Engines

	Rocketdyne 45,000 lb_f IME	Pratt & Whitney RL-10A-4	Russian D-57
Thrust	45,000 lb_f	20,800 lb_f	88,300 lb_f
Isp	467.5 s	448.9 s	456.5 s
Chamber Pressure	1195 psia	465 psia	1585 psia
Expansion Ration	160:1	85:1	143:1
Mass	825 lb_m	411 lb_m	1744 lb_m
Thrust/Weight Ratio	55:1	50.6:1	50.6:1
Notes	Paper design for National Launch System (NLS)	Atlas Centaur stage uses 2 engines based on Titan Centaur stage's RL-10A-3	Extensively tested but not flown; Aerojet seeking to market it

B.4.1.4 AVIONICS

The avionics is comprised of guidance and control, data handling, instrumentation, communications, flight termination, electrical power, and electrical harnesses. The suite is based on a detailed Aerospace upper stage design using state-of-the-art avionics based on today's computer technology. It is significantly lighter than the avionics in current vehicles.

B.4.1.5 MISCELLANEOUS

Additional miscellaneous components are based on Aerospace WERs.

B.4.1.6 PROPELLANT

The propellant consists of liquid hydrogen and oxygen. Residual propellant is calculated using Aerospace WERs to determine burn-out mass. In the case of the Titan stage, boil-off is also accounted for. The amount of usable propellant—23,507 kg (51,716 lb) for Atlas, 15,909 kg (35,000 lb) for Delta, and 20,454 kg (45,000 lb) minus boil-off for Titan—is then added to determine total stage mass.

B.4.2 Advanced Cryo Spacecraft

The spacecraft launched by advanced cryogenic upper stages are sized in the same manner as those of the baseline systems. Power is provided by advanced GaAs arrays, and propulsion is provided by chemical engines or hydrazine arcjets.

B.5 SOLAR ELECTRIC

The solar electric spacecraft are also very conventional. The general subsystems are sized as described above. A 3% additional shielding mass is added for GEO and MEO ORMs because of the long trip times through the Van Allen belts. These long trip times also force the inclusion of propellant residuals because of the in-transit losses (see previous discussion). Power is provided by flexible GaAs arrays that remain with the spacecraft throughout its life; the arrays power the transfer propulsion subsystem, which is comprised of arcjets, stationary plasma thrusters (SPTs), or ion engines. Electric propulsion orbit transfer is not used with the HEO ORM.

B.5.1 Power Subsystem

The power subsystem is based on advanced flexible GaAs arrays that are 21% efficient and have 245.84 W/m^2. They produce 61.1 W/kg. The arrays are an Advanced Photovoltaic Solar Array (APSA) derivative. Radiation protection is the equivalent of approximately 12 mil of top cover glass (the bottom substrate provides the equivalent of approximately 12-mil cover glass, the same as our rigid arrays). The deployment mechanism, consisting of a canister and a boom, has a mass of 50 kg. The same NiH_2 CVP battery, PMAD, and miscellaneous components used with the rigid arrays (sized for spacecraft requirements) round out the subsystem. The arrays are not jettisoned after transfer but remain with the spacecraft.

The power subsystem is sized according to the required transfer power by the iterative process described in the sidebar. Key is a set of curves derived from the solar array characteristics and output from the EVA program. EVA characterizes an "average" solar electric transfer system (since the results vary slightly depending on the specific transfer technology) for a 300-day GEO transfer. Figure B-1 shows trip time/burn time, BOL thrust/EOL thrust, and EOL power/BOL power as a function of starting altitude.

Sizing the Solar Electric Power Subsystem

Given a fixed trip time (determined by the ORM),

- Choose a drop-off altitude
- Determine the required transfer ΔV from the Edelbaum approximation for low thrust propulsion systems
- Determine the required burn time using the graph in Figure B-1 and the trip time
- Calculate average transfer thrust from the equation

$$(\text{Transfer Propellent Mass}) \times g_0 \times I_{sp} / (\text{Burn Time}) = \text{Average Thrust}$$

- Determine beginning-of-life (BOL) thrust from average thrust and the graph in Figure B-1
- Calculate BOL power from the equation

$$\{(\text{BOL Thrust}) / [2 \times (\text{Engine Efficiency}) \times (\text{PPU Efficiency})]\} \times I_{sp} \times g_0 = \text{BOL Power}$$

- Add housekeeping power, as determined from spacecraft sizing, to get total transfer power

Iterate the process by increasing payload requirements for the effectiveness analysis or by raising the initial orbit for the cost analysis until there is no more launch vehicle margin.

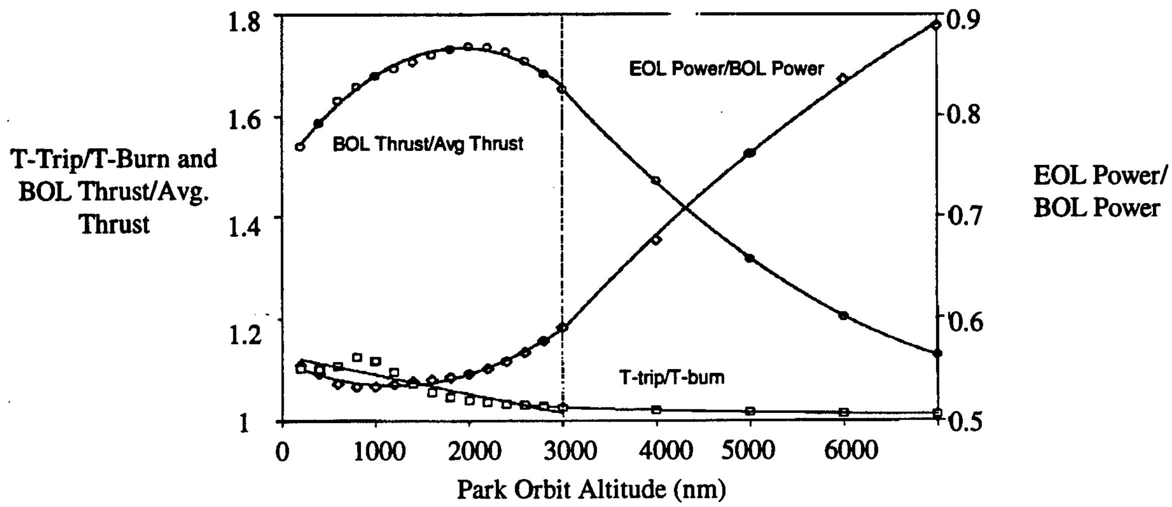


Figure B-1. Sizing curves for the solar electric system.

B.5.2 Propulsion Subsystem

The propulsion subsystem consists of either ammonia (NH_3) arcjets, hydrogen (H_2) arcjets, hydrazine (N_2H_4) arcjets, xenon (Xe) SPTs, or Xe ion engines for orbital transfer. On-orbit propulsion is provided by monopropellant N_2H_4 and/or the transfer thrusters, as summarized in Table B-8. Tanks are shared as much as possible within the spacecraft and, with the exception of the hydrogen tanks, are sized as before. The hydrogen tank consists of the following:

- The tank itself, based on propellant fraction and incorporating 3% ullage
- Secondary structure, based on a percentage of the hydrogen tank
- Thermal control (foam and MLI), whose thickness is based on historical data analyzed by Aerospace
- Plumbing, which is calculated as a percentage of the hydrogen tank
- Separation system, a simple clamp-based, spring-activated device designed by Aerospace, which has a constant mass

During the transfer, two electric propulsion units function at the same time; power conditioning is provided by a switching power processor to conserve mass. As mentioned before, the hydrogen arcjets and tanks separate from the spacecraft. For GEO, the spacecraft is brought to an altitude 300 nmi above GEO; separation occurs, and the spacecraft uses its on-orbit propulsion system to return to GEO. For MEO, the separation altitude is 50 nmi. A discussion of each propulsion subsystem option is given below. The thrusters are either based on existing thrusters chosen by Aerospace or are extrapolations of existing thrusters.

Table B-8. On-Orbit Thruster Options Considered With Nuclear and Solar Electric Lift

ORM	Stationkeeping	On-orbit Maneuver
GEO	8 monoprop N_2H_4 for E-W and 4 add'l transfer-type thrusters for N-S (N_2H_4 instead of H_2 arcjets)	Same electric thruster as S/K *or* Biprop $N_2H_4 + N_2O_4$
MEO	12 monoprop N_2H_4	Monoprop N_2H_4
LEO	12 monoprop N_2H_4 *or* 12 monoprop N_2H_4 and 4 add'l transfer-type thrusters (N_2H_4 not H_2 arcjets)	Monoprop N_2H_4 *or* Same electric thruster as S/K
HEO	N/A	N/A

B.5.2.1 AMMONIA ARCJETS

For orbital transfer, ammonia arcjets come in three sizes: (1) power less than 3500 W, (2) power between 3500 and 7000 W, and (3) power greater than 7000 W. These arcjets have been extrapolated from existing lower power arcjets. They have an I_{sp} of 800 s and a life of 2000 hr (83 days), meaning that some missions (e.g., GEO) must carry multiple sets of arcjets. The power conditioning unit is sized for the thruster.

For on-orbit propulsion, the ammonia arcjets are 2600 W and have an I_{sp} of 700 s. Power conditioning is sized for the thruster, and the NH_3 tank is shared with the transfer arcjets. Plumbing is sized as before.

B.5.2.2 HYDROGEN ARCJETS

For orbital transfer, hydrogen arcjets come in the same three sizes as the ammonia arcjets. Their I_{sp} is 1200 s, and they have a life of 2000 hr (83 days). The power conditioning unit is sized for the thruster. Hydrogen is not used for on-orbit propulsion due to storage concerns.

B.5.2.3 HYDRAZINE ARCJETS

For orbital transfer, hydrazine arcjets come in the same three sizes as the ammonia arcjets. Their I_{sp} is 550 s, and they have a life of 2000 hr (83 days). The power conditioning unit is sized for the thruster. The thruster is only used for LEO missions.

For on-orbit propulsion, the thrusters are identical to the hydrazine thrusters of the baseline chemical system, which have already been discussed. The subsystem is sized as before. The tanks are shared whenever possible.

B.5.2.4 XENON SPT

For orbital transfer, the Xe SPTs come in the same three sizes as the ammonia arcjets. They have an I_{sp} of 1600 s and a life of 10,000 hr (417 days). Only one set of SPTs are used on the spacecraft (that is, there is no redundancy). Power conditioning is sized for the thruster.

For on-orbit propulsion, the thrusters are 1350 W and have an I_{sp} of 1600 s. Power conditioning is based on the thruster requirements, plumbing is sized as before, and the Xe tank is shared with the transfer thrusters.

B.5.2.5 XENON ION ENGINES

For orbital transfer, the Xe ion engines come in the same three sizes as the ammonia arcjets. I_{sp} is 3200 s, and life is 10,000 hr (417 days). Only one set of ion engines is carried on the spacecraft for transfer. Power conditioning is sized for the specific thruster.

For on-orbit propulsion, the thrusters are based on NASA-Lewis Research Center's (LeRC) 500-W, 30-cm engines with an I_{sp} of 3200 s. Power conditioning is based on thruster requirements, plumbing is as before, and the Xe tank is shared with the transfer thrusters.

B.6 NUCLEAR ELECTRIC

The nuclear electric spacecraft is similar to the solar electric spacecraft except the transfer and on-orbit power is provided by a nuclear reactor rather than solar arrays. The general subsystems are sized as described above. A 3% shielding mass is added to the normal structural mass for the GEO and MEO ORMs due to long trip times through the Van Allen belts. The propulsion subsystems are identical to those of the solar electric systems described above.

The characteristics of the nuclear power subsystem are based on historical information (SP-100, S-Prime, Topaz) analyzed by Rocketdyne. Reactor power is sized from transfer power requirements. The methodology is similar to that used with solar electric except

- Burn time equal trip time (since solar eclipsing is not a factor)
- Thrust is constant
- Thrust drives power requirements

Given power requirements, the mass of the nuclear subsystem is determined from the curve in Figure B-2. Using relationships from typical systems, the various components are then broken out as a percentage of the total. Components include

- Reactor
- Radiation shielding
- Boom and structure
- Heat rejection
- Power conditioning and control (for the reactor)

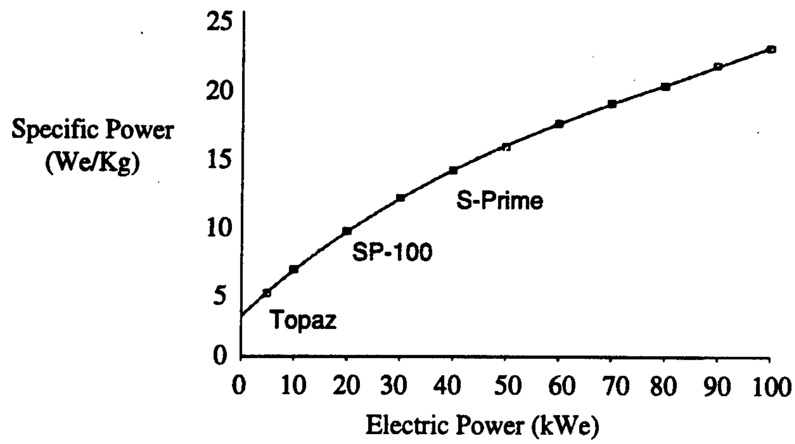


Figure B-2. Nuclear electric reactor performance.

The PMAD system is sized as before with the photovoltaic power systems. The system does not have a battery since eclipsing is not a concern for nuclear systems. Also, all thrusters and tanks remain with the spacecraft (i.e., hydrogen arcjets do not separate).

B.7 NUCLEAR BIMODAL

Nuclear bimodal spacecraft derive their power and primary propulsion from a single nuclear reactor. Propulsion is provided by passing a gas (hydrogen or ammonia) through channels in the reactor. The system is also designed to convert reactor heat into electricity for the spacecraft. The general subsystems of the spacecraft are sized as before. The reactor and hydrogen tank remain with the spacecraft through its life. The bimodal system used in the OECS is based on Phillips Lab and DOE's NEBA-1 Concept 3 design.

B.7.1 Power Subsystem

The power subsystem centers around the nuclear bimodal system. The NEBA-1 Concept 3 is a fixed design consisting of reactor, radiation shield, heat transport, heat rejection, power conversion, wiring harness (near the reactor), boom and other structure, and power conversion (multicouple thermoelectric diodes). Certain elements of the design are scaled by OCEM based on specific spacecraft power requirements (e.g., power conversion and wiring harnesses).

The remaining elements of the spacecraft power subsystem are batteries, P&PS controller, and spacecraft power cabling. These items are sized as they were sized for photovoltaics.

B.7.2 Propulsion Subsystem

Orbital transfer propulsion is provided by the nuclear bimodal system using hydrogen as the propellant. The single reactor design of the NEBA-1 Concept 3 produces 2200 N thrust with an I_{sp} of 820 s. Unlike the electric propulsion and solar thermal/bimodal systems, where thrust is determined by the trip time, the fixed thrust determines the trip time. The actual trip time and required ΔV for a given drop-off altitude and launch vehicle was determined by Rocketdyne. The hydrogen tank, which is sized in the manner described above for solar electric hydrogen arcjets, does not separate from the system upon reaching final orbit.

On-orbit propulsion options are summarized in Table B-9. If monopropellant hydrazine and hydrazine arcjets are used, they are sized as described before. When the nuclear bimodal system is also used (for high power GEO maneuvers), ammonia is used as the propellant. An ammonia tank is sized with the system using the same method applied with solar electric ammonia arcjets.

Table B-9.On-Orbit Thruster Options Considered With Nuclear Bimodal Lift

ORM	Stationkeeping	On-orbit maneuver
GEO	8 monoprop N_2H_4 for E-W and 4 N_2H_4 arcjets for N-S	N_2H_4 arcjet *or* Monoprop N_2H_4 *or* NH_3 nuclear bimodal
MEO	12 monoprop N_2H_4	Monoprop N_2H_4
LEO	NH_3 nuclear bimodal and 8 monoprop N_2H_4	Same as stationkeeping
HEO	4 N_2H_4 arcjets and 8 monoprop N_2H_4 *or* 12 monoprop N_2H_4	Same as stationkeeping

B.8 SOLAR THERMAL

The spacecraft using solar thermal technologies is very similar to the baseline chemical spacecraft. The general subsystems are sized as previously described, and power is provided by the same rigid GaAs photovoltaic solar arrays (sized to meet requirements). Transfer propulsion, however, is provided by a solar thermal system, which concentrates sunlight onto a receiver through which hydrogen propellant passes. The solar thermal system separates from the spacecraft upon reaching final orbit. GEO and MEO are handled the same way as the solar electric hydrogen arcjets; HEO is dropped off in its orbit.

Orbital transfer propulsion is provided by the solar thermal systems, which consist of two inflatable mirrors on a rigid structure (the reflectors), the inflation mechanism, turntables and junctures (for mirror control), and one thruster/receiver. A hydrogen tank and separation system round out the transfer propulsion system. Guidance and control is provided by the spacecraft. The sizing relationships for the solar thermal components were provided by Phillips Lab (PL/RK). Hydrogen tank sizing is as before, and the separation system is the previously mentioned Aerospace-designed component.

Sizing the orbit transfer system involves an iterative process. The trip time is fixed by the ORM. Given an initial drop-off altitude, the transfer ΔV and I_{sp} are determined from Rocketdyne's analysis described in Section 4.3.9. Since the initial thrust-to-weight ratio is constant for all cases (an OECS simplifying assumption), the thrust is derived from the initial sizing mass, which also determines BOL power. The thruster and reflector mass is then read off a curve as a function of power based on PL/RK data. The process is repeated, either raising initial altitude (for the cost analysis) or increasing payload requirements (for the effectiveness analysis), until the selected launch vehicle has zero launch margin.

On-orbit propulsion options are summarized in Table B-10. These contain the same thrusters previously discussed.

Table B-10. On-Orbit Thruster Options Considered With Solar Thermal Lift

ORM	Stationkeeping	On-orbit maneuver
GEO	8 monoprop N_2H_4 for E-W and 4 N_2H_4 arcjets for N-S	N_2H_4 arcjet *or* Monoprop N_2H_4
MEO	12 monoprop N_2H_4	Monoprop N_2H_4
LEO	12 monoprop N_2H_4 *or* 4 NH_3 arcjet, N_2H_4 arcjet, Xe SPT, or Xe ion and 8 monoprop N_2H_4	Same as stationkeeping
HEO	4 N_2H_4 arcjet and 8 monoprop N_2H_4 *or* 12 monoprop N_2H_4	Same as stationkeeping

B.9 SOLAR BIMODAL

Like nuclear bimodal, solar bimodal provides both power and propulsion to the spacecraft. All other aspects of the spacecraft are identical to the systems previously discussed.

B.9.1 Power Subsystem

The power subsystem centers around the solar bimodal system. This system consists of

- Two rigid collectors (mirrors)
- A receiver, which contains graphite for thermal energy storage, the thruster, the cavity to heat the propellant, room for thermal energy conversion diodes, and heat rejection
- Diodes for thermal energy conversion

The collectors and receiver are sized for the maximum of power or propulsion energy requirements (propulsion requirements typically drive the system). The diodes are sized based on power requirements only. The power subsystem also contains a rigid 1-m^2 GaAs solar array and a battery (both are identical to those used before). The PMAD, as discussed before, rounds out the spacecraft's power subsystem.

B.9.2 Propulsion Subsystem

Orbital transfer is provided by the bimodal system. Transfer requirements typically size the collectors and receiver of the system. The process is iterative: the trip time is fixed based on the ORM. Transfer ΔV, I_{sp}, initial thrust-to-weight, and solar bimodal power are determined from

Rocketdyne's analysis once an initial orbit is selected (Section 4.3.9). The rest of the spacecraft is sized and the process iterated until there is no launch vehicle margin.

Hydrogen is used as the transfer propellant. The tank is sized as previously discussed. The tank remains with the spacecraft throughout its life.

Table B-11 summarizes the solar bimodal spacecraft on-orbit propulsion options. Most options are identical to those previously discussed. Like nuclear bimodal, when solar bimodal is an on-orbit option, ammonia is used as the propellant.

Table B-11. On-Orbit Thruster Options Considered With Solar Bimodal Lift

ORM	Stationkeeping	On-orbit Maneuver
GEO	8 monoprop N_2H_4 for E-W and 4 N_2H_4 arcjets for N-S	N_2H_4 arcjet *or* Monoprop N_2H_4 *or* NH_3 solar bimodal
MEO	12 monoprop N_2H_4	Monoprop N_2H_4
LEO	NH_3 solar bimodal and 8 monoprop N_2H_4	Same as stationkeeping
HEO	NH_3 solar bimodal and 8 monoprop N_2H_4 *or* 12 monoprop N_2H_4	Same as stationkeeping

APPENDIX C

RSM EQUATIONS FOR AVAILABILITY AND NUMBER OF SATELLITES BOUGHT

This appendix contains the response surface methodology (RSM) coefficients for maximum constellation availability and corresponding number of satellites bought for each ORM. Data for all ORMs except ORM 2a (MEO-GPS) was fit using five-factor, central composite, face-centered designs consisting of 32 points: 10 face-centered points, 16 corner points, and 6 center points. ORM 2a (GPS) was fit with a five-factor D-optimal design.

In each case the five factors were:

- Number of satellites (N_S)
- Satellite mean mission duration (M_{MD})
- Satellite deployment time (T_D)
- Launch reliability (R_L)
- Minimum time between launches (T_B)

Table C-1 lists the ranges for each ORM over which the five parameters were fit. The equations should not be used outside these ranges. Table C-2 through Table C-5 contain the coefficients for ORM 1 (GEO), ORM 2a (MEO-GPS), ORM 3a (LEO-polar), and ORM 4 (HEO), respectively. In each table the first column specifies the term and the second column the transformed term to which the coefficients apply. The remaining columns contain the coefficients of the equations for availability and number of satellites bought for either one or two values of on-orbit spares. Missing coefficients correspond to terms eliminated in the multiple linear regression fitting process. The R-squared-adjusted and the RSM fit error are given at the bottom of each table.

Table C-1. Applicable Parameter Ranges for the RSM Approximations

Parameter	ORM 1 (GEO)	ORM 2a (GPS)	ORM 3a (LEO-polar)	ORM 4 (HEO)
N_S[a]	3–5	12–21	1–3	1–3
M_{MD} (yr)	5–14	8–14	3–8	5–14
T_D (day)	60–360	60–270	60–90	60–120
R_L	0.87–0.98	0.87–0.98	0.87–0.98	0.87–0.98
T_B (yr)	0.125–0.333	0.125–0.25	0.125–0.333	0.125–0.333

[a]Constellation size is without active spares

Table C-2. RSM Availability Coefficients for ORM 1 (GEO)

Term	Transformed Term	Availability With 0 Spares	Availability With 1 Spare	No. Bought With 0 Spares	No. Bought With 1 Spare
1		0.901341	0.992455	3.071515	3.439753
N_S	(N_S-4)	-0.027000	-0.007134	0.404726	0.373488
M_{MD}	$(M_{MD}-9.5)/4.5$	0.048821	0.011917	-0.709443	-0.793207
T_D	$(T_D-210)/150$	-0.070878	-0.013927	-0.041533	-0.047038
R_L	$(R_L-0.925)/0.055$	0.015790	0.005172	-0.084309	-0.088456
T_B	$(T_B-0.229)/0.104$	-0.005063	-0.002605	0.008079	0.018500
$N_S \times N_S$	$(N_S-4)^2$				
$M_{MD} \times M_{MD}$	$[(M_{MD}-9.5)/4.5]^2$	-0.020612	-0.006504	0.057232	0.083799
$T_D \times T_D$	$[(T_D-210)/150]^2$		-0.004504		
$R_L \times R_L$	$[(R_L-0.925)/0.055]^2$				
$T_B \times T_B$	$[(T_B-0.229)/0.104]^2$				
$N_S \times M_{MD}$	$(N_S-4)\times(M_{MD}-9.5)/4.5$	0.008187	0.004625	-0.092867	-0.072139
$N_S \times T_D$	$(N_S-4)\times(T_D-210)/150$	-0.016187	-0.006125	-0.003435	-0.007766
$N_S \times R_L$	$(N_S-4)\times(R_L-0.925)/0.055$		0.002295	-0.010775	-0.012703
$N_S \times T_B$	$(N_S-4)\times(T_B-0.229)/0.104$	-0.005063	-0.003250		0.013562
$M_{MD} \times T_D$	$(M_{MD}-9.5)/4.5\times(T_D-210)/150$	0.028063	0.010375	0.018477	0.026438
$M_{MD} \times R_L$	$(M_{MD}-9.5)/4.5\times(R_L-0.925)/0.055$	-0.004289	-0.002769	0.011386	0.017952
$M_{MD} \times T_B$	$(M_{MD}-9.5)/4.5\times(T_B-0.229)/0.104$	0.005937	0.002500	-0.008207	0.010961
$T_D \times R_L$	$(T_D-210)/150\times(R_L-0.925)/0.055$	0.009943	0.003813		0.016048
$T_D \times T_B$	$(T_D-210)/150\times(T_B-0.229)/0.104$				
$R_L \times T_B$	$(R_L-0.925)/0.055\times(T_B-0.229)/0.104$	0.006267	0.004911		
R-sq-adj		**0.9918**	**0.9695**	**0.9998**	**0.9974**
RMS Error		**0.006853**	**0.00349**	**0.01084**	**0.00344**

Table C-3. RSM Availability Coefficients for ORM 2a (GPS)

Term	Transformed Term	Availability With 3 Spares	No. Bought With 3 Spares
1		0.992804	45.684739
N_S	(NS-16.5)/4.5		11.253998
M_{MD}	(MMD-11)/3	0.005101	-8.837363
T_D	(TD-165)/105	-0.005210	-0.521043
R_L	(R_L-0.925)/0.055	0.009765	-2.038741
T_B	(TB-0.1875)/0.0625	-0.003201	1.479615
$N_S \times N_S$	$[(NS-16.5)/4.5]^2$		
$M_{MD} \times M_{MD}$	$[(MMD-11)/3]^2$	-0.002198	-1.718261
$T_D \times T_D$	$[(TD-165)/105]^2$	-0.002302	
$R_L \times R_L$	$[(R_L-0.925)/0.055]^2$		
$T_B \times T_B$	$[(TB-0.1875)/0.0625]^2$		
$N_S \times M_{MD}$	(NS-16.5)/4.5×(MMD-11)/3		-1.704751
$N_S \times T_D$	(NS-16.5)/4.5(TD-165)/105		
$N_S \times R_L$	(NS-16.5)/4.5×(R_L-0.925)/0.055		-0.304593
$N_S \times T_B$	(NS-16.5)/4.5×(TB-0.1875)/0.0625		0.479025
$M_{MD} \times T_D$	(MMD-11)/3×(TD-165)/105	0.002386	0.341815
$M_{MD} \times R_L$	(MMD-11)/3×(R_L-0.925)/0.055	-0.002281	0.570239
$M_{MD} \times T_B$	(MMD-11)/3×(TB-0.1875)/0.0625	0.001196	0.522119
$T_D \times R_L$	(TD-165)/105×(R_L-0.925)/0.055	0.001256	0.215889
$T_D \times T_B$	(TD-165)/105×(TB-0.1875)/0.0625	-0.003204	
$R_L \times T_B$	(R_L-0.925)/0.055×(TB-0.1875)/0.0625	0.002367	
R-sq-adj		**0.9712**	**0.9993**
RMS Error		**0.0188**	**0.1574**

Table C-4. RSM Availability Coefficients for ORM 3a (LEO-Polar)

Term	Transformed Term	Availability With 0 Spares	Availability With 1 Spare	No. Bought With 0 Spares	No. Bought With 1 Spare
1		0.935214	0.998544	2.681363	3.289157
N_S	(N_S-2)	-0.042378	-0.002681	0.753861	0.602371
M_{MD}	$(M_{MD}-5.5)/2.5$	0.034847	0.002244	-0.579568	-0.726983
T_D	$(T_D-75)/15$	-0.013204	-0.001161	-0.076459	
R_L	$(R_L-0.925)/0.055$	0.012614	0.001736		-0.093657
T_B	$(T_B-0.229)/0.104$	-0.011195	-0.001653		
$N_S \times N_S$	$(N_S-2)^2$		-0.001003		
$M_{MD} \times M_{MD}$	$[(M_{MD}-5.5)/2.5]^2$	-0.015771	-0.001003	0.103812	0.194645
$T_D \times T_D$	$[(T_D-75)/15]^2$				
$R_L \times R_L$	$[(R_L-0.925)/0.055]^2$				
$T_B \times T_B$	$[(T_B-0.229)/0.104]^2$	0.004475			
$N_S \times M_{MD}$	$(NS-2)\times(M_{MD}-5.5)/2.5$	0.017593	0.001875	-0.158200	-0.122735
$N_S \times T_D$	$(NS-2)\times(T_D-75)/15$	-0.006505	-0.001000		
$N_S \times R_L$	$(NS-2)\times(R_L-0.925)/0.055$	0.0006222	0.001381	-0.019411	-0.018163
$N_S \times T_B$	$(NS-2)\times(T_B-0.229)/0.104$	-0.008966	-0.001500		
$M_{MD} \times T_D$	$(M_{MD}-5.5)/2.5\times(T_D-75)/15$	0.007950	0.001375		
$M_{MD} \times R_L$	$(M_{MD}-5.5)/2.5\times(R_L-0.925)/0.055$	-0.004276	-0.001012	0.012109	0.016649
$M_{MD} \times T_B$	$(M_{MD}-5.5)/2.5\times(T_B-0.229)/0.104$	0.005138	0.001125		
$T_D \times R_L$	$(T_D-75)/15\times(R_L-0.925)/0.055$		0.001133		
$T_D \times T_B$	$(T_D-75)/15\times(T_B-0.229)/0.104$	0.004662	-0.000750		
$R_L \times T_B$	$(R_L-0.925)/0.055\times(T_B-0.229)/0.104$	0.006599	0.001384		
R-sq-adj		**0.9917**	**0.9580**	**0.9978**	**0.9991**
RMS Error		**0.004516**	**0.000819**	**0.02877**	**0.02202**

Table C-5. RSM Availability Coefficients for ORM 4 (HEO)

Term	Transformed Term	Availability With 0 Spares	Availability With 1 Spare	No. Bought With 0 Spares	No. Bought With 1 Spare
1		0.976328	0.999226	4.739343	0.374384
N_S	(N_S-2)	-0.014750	-0.001251	2.658980	-0.068209
M_{MD}	$(M_{MD}-9.5)/4.5$	0.013445	0.001032	-2.323438	0.091559
T_D	$(T_D-90)/30$	-0.008147	-0.000598	-0.035434	0.001437
R_L	$(R_L-0.925)/0.055$	0.004500	0.000861	-0.283157	0.011124
T_B	$(T_B-0.229)/0.104$	-0.002364	-0.000607	0.000934	-0.000283
$N_S \times N_S$	$(N_S-2)^2$				0.013869
$M_{MD} \times M_{MD}$	$[(M_{MD}-9.5)/4.5]^2$	-0.006696	-0.000497	0.568446	0.011978
$T_D \times T_D$	$[(T_D-210)/150]^2$				
$R_L \times R_L$	$[(R_L-0.925)/0.055]^2$		0.000671		
$T_B \times T_B$	$[(T_B-0.229)/0.104]^2$				
$N_S \times M_{MD}$	$(N_S-2)\times(M_{MD}-9.5)/4.5$	0.006538	0.000781	-1.177110	-0.014978
$N_S \times T_D$	$(N_S-2)\times(T_D-90)/30$	-0.004588	-0.000519		
$N_S \times R_L$	$(N_S-2)\times(R_L-0.925)/0.055$	0.002192	0.000649	-0.14679	-0.001449
$N_S \times T_B$	$(N_S-2)\times(T_B-0.229)/0.104$	-0.001750	-0.000519		
$M_{MD} \times T_D$	$(M_{MD}-9.5)/4.5\times(T_D-90)/30$	0.004314	0.000681		
$M_{MD} \times R_L$	$(M_{MD}-9.5)/4.5\times(R_L-0.925)/0.055$	-0.001167	-0.000410	0.103382	0.002633
$M_{MD} \times T_B$	$(M_{MD}-9.5)/4.5\times(T_B-0.229)/0.104$	0.001750	0.000306		
$T_D \times R_L$	$(T_D-90)/30\times(R_L-0.925)/0.055$	0.001347	0.000320		
$T_D \times T_B$	$(T_D-90)/30\times(T_B-0.229)/0.104$		-0.000344	0.053467	-0.000758
$R_L \times T_B$	$(R_L-0.925)/0.055\times(T_B-0.229)/0.104$	0.003027	0.000671		
R-sq-adj		**0.9891**	**0.9805**	**0.9988**	**0.9999**
RMS Error		**0.001958**	**0.000286**	**0.1006**	**0.001016**

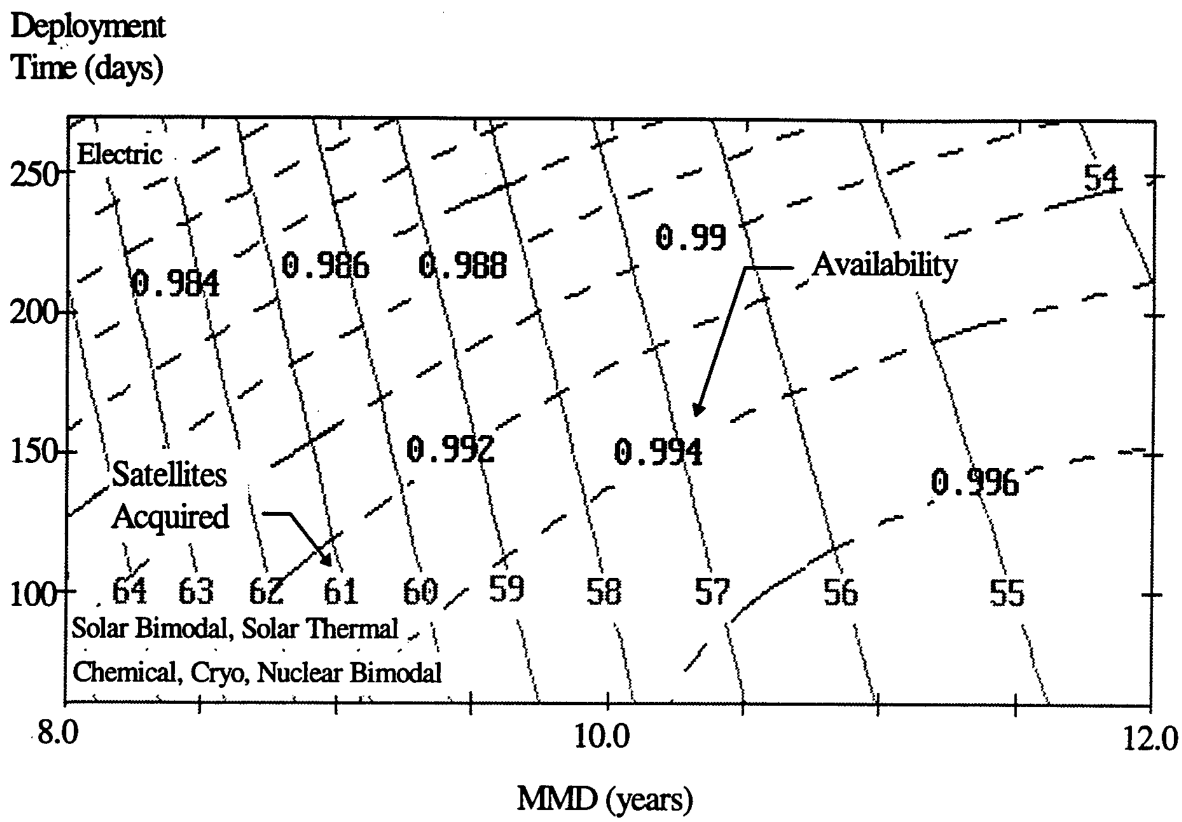


Figure C-1. ORM 2a (GPS): Availability and number of satellites required to maintain a 21-satellite constellation with 3 spares.

ORM 2 (GPS) represents a different situation from ORM 1 (GEO) in that the constellation was designed with three active on-orbit spares. We see from Figure C-1 that, with the three active spares, high availabilities are still achieved when electric propulsion is used despite the long deployment times. No cases using additional spares were examined for ORM 2 (GPS).

APPENDIX D

INNOVATIVE TECHNOLOGY VOLUME CALCULATIONS

D.1 INTRODUCTION

This appendix provides a description of the methodologies used to estimate the volumes of the satellite payload and bus, the lift propulsion propellant tank, and other innovative lift propulsion elements in the OCEM design and sizing model. The effectiveness analysis (Chapter 5) compares the resulting integrated volumes against current launch-vehicle fairing volumes to identify potential technology- and launch vehicle-specific volumetric constraints.

All payload and bus volumes are estimated based on a constant wet density of 79 kg/m^3 (Larsen and Wertz, p. 292) except the following lift propulsion-specific items, which cannot be considered standard bus hardware:

- Nuclear bimodal: lift propellant tank and nuclear bimodal system (including nozzle)
- Solar bimodal: lift propellant tank, collectors, and receiver/absorber/converter (including thruster)
- Solar thermal: lift propellant tank, collectors, and thruster
- Nuclear electric: lift propellant tank and nuclear power system
- Solar electric: lift propellant tank and lift propulsion solar arrays

The volume of the lift propellant tanks is calculated by OCEM based on propellant needs (see Appendix B). The methodologies for the other elements are presented below. We have determined that an unusable volume is associated with all of the above nontank elements. This unusable volume is space surrounding the component that is not needed for lift propulsion hardware. The space is located too far from the payload and bus to provide them with additional unusable volume. For each of these elements, except the solar arrays, we estimate its unusable volume and combine it with the actual volume to arrive at an effective volume. The unusable volume associated with solar arrays is believed to be small and is ignored. Volume implications of the advanced cryogenic upper stages are also discussed below. A summary of the methodologies appears in Table D-1.

D.2 ADVANCED CYROGENIC

Except for Titan IV, the advanced cryogenic upper stages do not intrude into the fairing volume. The advanced cryogenic stage for Delta II replaces the current Delta second stage. For Atlas IIAS, the advanced cryogenic stage replaces the Centaur. However, because the advanced cryogenic stage design for Titan IV is larger than the Titan Centaur it replaces, an additional 1 m at the bottom of the Titan fairing is occupied by the advanced stage.

Table D-1. Summary of the Assumptions Used to Calculate Required Fairing Volumes

Innovative Technology	Payload (kg/m^3)	Bus (kg/m^3)	Propellant Tank Volume	Lift Technology Volume
Cryogenic	79	79	Fixed. Except for Titan IV, tanks are not contained in fairing. All are of fixed volume.	Delta and Atlas fairings unaffected. Titan fairing loses 1 m in usable length.
Nuclear Bimodal	79	79	Case specific. H_2 tank diameter = diameter of fairing dynamic envelope.	Reactor placed at top of fairing. Volume based on length of fairing required (see Figure D-1). Space around reactor is considered unusable.
Solar Bimodal	79	79	Case specific. H_2 tank diameter reduced by stowed collectors.	Collectors stowed in cylindrical shell around H_2 tank and receiver/absorber/collector (see Figure D-2). Space between receiver/absorber/collector and shell considered unusable(see Figure D-3). Shell thickness, thus volume, a function of thermal power.
Solar Thermal	79	79	Case specific. Tank diameter = diameter of fairing dynamic envelope.	Collectors and thruster packaged in a disk of height *h*, which is added to height of H_2 tank (Figure D-4). Unused space in disk is considered unusable.
Nuclear Electric	79	79	Case specific. H_2 tank diameter = diameter of fairing dynamic envelope. NH3 and Xe tanks treated only as volumes.	Nuclear system placed at top of fairing. Volume based on system length (see Figure D-1), which is a function of electrical power. Space around reactor is considered unusable.
Solar Electric	79	79	Case specific. H_2 tank diameter = diameter of fairing dynamic envelope. NH3 and Xe tanks treated only as volumes.	Solar array volume based on 15 kW/m^3 of BOL power.

D.3 NUCLEAR BIMODAL AND NUCLEAR ELECTRIC

For the nuclear bimodal and nuclear electric systems, we must determine not only the system's volume (reactor, reactor shield, thermal radiator, and the telescoping boom) but any associated unusable volume. We have assumed the nuclear system is placed at the top of the fairing above the hydrogen tank. (For NH_3 and Xe, the propellant tanks are much smaller and their volumes are simply combined with the other volumes.) Thus the payload and remaining satellite bus are at the bottom of the fairing. We further assume that the nuclear system, which has an approximately cylindrical shape, is oriented as shown in Figure D-1. All nonreactor space around the system's longitudinal axis is regarded as unusable and part of the reactor's effective volume.

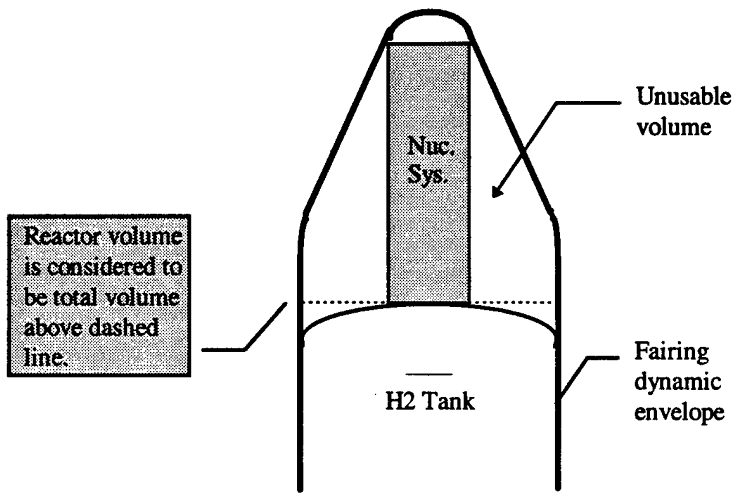


Figure D-1. Assumed placement of the nuclear system for determining required fairing volume.

The length and diameter of the bimodal reactor cylinder are a constant 4.74 m and 1.26 m, respectively, based on the NEBA 1 design (Vanek). In the nuclear electric case, length and diameter are functions of the electric power as shown in Table D-2 (Determan). Since each launch vehicle's fairing is a different size and shape, effective nuclear system volume as a function of electric power is different for each. Estimated effective volumes as a function of power are shown in Table D-3. The linear least square fits of effective volume in m^3 for each launch vehicle are:

- Delta II: $20.29 + 0.43x$
- Atlas IIAS: $12.68 + 0.75x$
- Titan IV: $39.72 + 1.23x$

where x is specific power in kWe. Standard errors of the fits are in the range of 1–5% of the total fairing volumes.

Table D-2. Maximum Length and Diameter of a Nuclear System

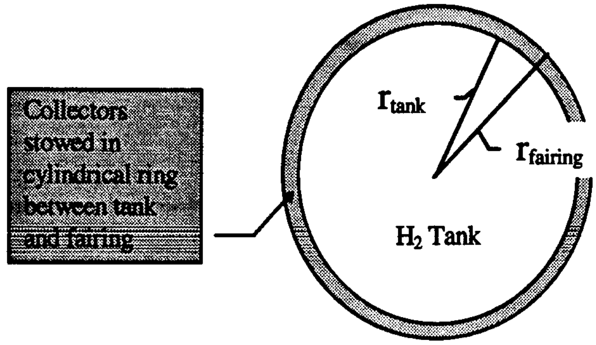
Electric Power (kW)	Length (m)	Diameter (m)
5.5	4.3	1.67
20	5.60	2.17
40	6.78	2.63

Table D-3. Estimated Required Volume for a Nuclear Electric System

Electric Power (kw)	Delta II (m^3)	Atlas IIAS (m^3)	Titan I (m^3)
5.5	22	16	45
20	30	29	67
40	37	42	88

D.4 SOLAR BIMODAL

For the solar bimodal case, we must consider the effective volume occupied by the collectors and the receiver/absorber/converter. The tabular data appearing in this section were provided for the OECS by Phillips Laboratory VTP (Malloy). We assume that the solar bimodal propulsion is at the bottom of the fairing below the hydrogen tank. The collectors are stored in an annulus between the hydrogen tank and the fairing as depicted in Figure D-2. The formula for the cross-sectional area of the annulus is indicated in the figure. The volume of annulus is found by multiplying this area by the stowed collector length. The thickness of the annulus is given in Table D-4 for each launch vehicle as a function of the system's maximum achievable electrical power. The stowed collector length is found in Table D-4 as well. Note that because of changes in the way the collectors are hinged, neither annular thickness nor collector length increase monotonically with power.



$\Delta r = r_{fairing} - r_{tank}$

Annulus area $= \pi \cdot \Delta r \cdot (r_{fairing} + r_{tank})$

Cross Section Through Fairing, Collectors, and H_2 Tank

Figure D-2. Cross-section of the annular area used to store the collector in a solar bimodal system.

Table D-4. Collector Length and Annular Thickness for a Solar Bimodal System[a,b]

Maximum Achievable Electric Power (kW)	Stowed Length (m)	Thickness (m)		
		Delta II	Atlas IIAS	Titan IV
1.0	4.2	0.08	0.06	0.04
2.0	2.9	0.16	0.12	0.09
3.0	3.6	0.25	0.19	0.13
5.0	4.5	0.14	0.10	0.07
7.5	5.8	0.22	0.16	0.11
10.0	6.5	0.30	0.22	0.15

[a]Four hinge lines used on 5kW and larger collectors.
[b]Optional horizontal hinge line used on 2 kW and larger collectors.

The solar bimodal receiver/absorber/converter sits at the bottom of the stack beneath the H_2 tank (see Figure D-3). We have assumed that, because of its location, the volume around it is unusable. Therefore the effective volume of this element is the product of its height and the area of the inner circle defined by the stowed collectors. The height of the receiver/absorber/collector as a function of rated electric power is given in Table D-5.

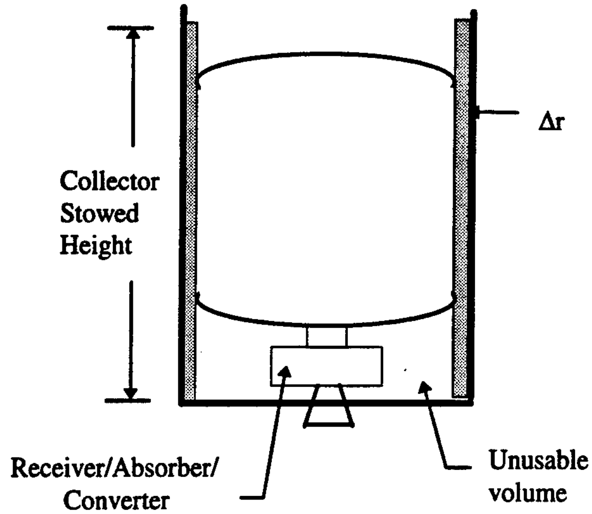


Figure D-3. Longitudinal cross section of fairing showing placement of the propulsion elements in a solar bimodal system.

Table D-5. Height of Solar Bimodal Receiver/Absorber/Converter

Rated Electrical Power (kW)	Receiver/Absorber/ Converter Height (m)
1.0	0.59
2.0	0.77
3.0	0.81
5.0	0.92
7.5	1.43
10.0	1.54

D.5 SOLAR THERMAL

Solar thermal packaging places the collectors and thruster in a cylindrical volume of height, h, and diameter equal to the fairing diameter (Perkins). The arrangement is shown in two views in Figure D-4. The estimated values of h are given in Table D-6 (Perkins). Because of the location of these elements at the bottom of the hydrogen tank, the volume around the collectors and thruster are regarded as unusable. Thus the effective volume of these elements is the entire volume of the cylindrical wafer ($\pi r^2 h$). For NH^3 and Xe, the propellant tanks are much smaller and their volumes are simply combined with the other volumes.

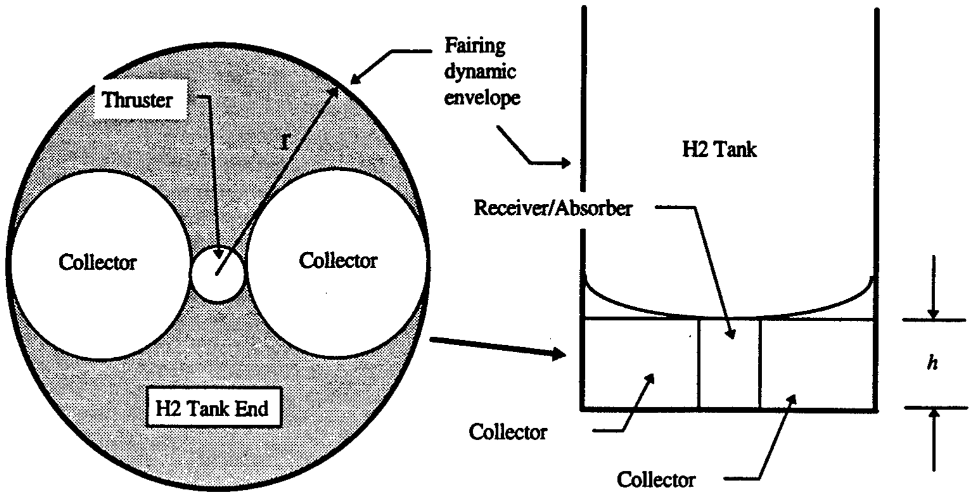


Figure D-4. Plan and longitudinal cross sections of fairing showing the solar thermal propulsion elements of a solar thermal system.

Table D-6. Additional Length of H_2 Tank, *h*, in the Solar Thermal System

Power (kW)	Delta II (m)	Atlas IIAS (m)	Titan IV (m)
20	0.25	0.18	0.15
50	0.38	0.28	0.23
100	0.56	0.41	0.33
200	0.81	0.58	0.46
400	1.17	0.84	0.66
500	1.32	0.94	0.74
1 MW	2.01	1.40	1.07
2	3.15	2.13	1.60
4	N/A[a]	3.40	2.49
8	N/A[a]	N/A[a]	4.01

[a] Greater than fairing length.

D.6 SOLAR ELECTRIC

The volume of the solar panels for the solar electric lift propulsion, including deployment mechanisms, is based on a power density of 15 kW/m^3. This estimate is based on an Advanced Photovoltaic Solar Array (APSA) derivative power density of 13.9 kW/m^3 for similar but 3% less efficient arrays (Gledhill and Marvin).

APPENDIX E
ACQUISITION COST ESTIMATES

We dealt with relative costs of the innovative technologies in the cost-effectiveness discussions in Chapter 7 to provide the most appropriate cost comparisons within the context of our CERs. In this appendix, we provide the actual cost estimates of establishing and maintaining constellations for 15 years in ORM 1 (GEO), ORM 2a (GPS), ORM 3a (LEO), and ORM 4 (HEO).

Table E-1. ORM 1 (GEO): Estimated Cost of Establishing and Maintaining Constellations for 15 Years

Payload Mass (kg)	Total # Sat.	Baseline		Advanced Cryo		Nuclear Bimodal			Solar Bimodal			Solar Thermal		Nuclear Electric	
	Lift	Chem	Chem	Cryo	Cryo	H_2	H_2	H_2	H_2	H_2	H_2	H_2	H_2	NH_3 Arc	NH_3 Arc
	Hold	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	NH_3 Arc	NH_3 Arc
	Move	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4	Chem	NH_3	N_2H_4	Chem	NH_3	N_2H_4 Arc	Chem	NH_3 Arc	Chem
		Cost ($B FY 95) per Constellation Using Specified Launch Vehicle[a] (MMD/sat. = 10 yr, num. moves/sat. = 3.25)													
		Payload Power = 0.50 kW													
200	5	0.74 D	0.74 D	1.16 D	1.17 D	2.39 A	2.39 A	2.39 A	0.81 L	0.82 L	0.81 L	0.76 L	0.76 L	3.21 T	3.2 T
	12	1.62 D	1.62 D	2.10 D	2.10 D	3.92 A	3.92 A	3.93 A	1.53 L	1.54 L	1.54 L	1.45 L	1.45 L	5.55 T	5.5 T
	18	2.27 D	2.27 D	2.81 D	2.81 D	5.04 A	5.04 A	5.05 A	2.02 L	2.04 L	2.03 L	1.92 L	1.93 L	7.38 T	7.4 T
	25	2.99 D	2.99 D	3.59 D	3.59 D	6.30 A	6.30 A	6.31 A	2.56 L	2.59 L	2.58 L	2.44 L	2.45 L	9.46 T	9.4 T
800	5	1.34 A	1.34 A	1.68 A	1.68 A	2.77 A	2.77 A	2.78 A	1.55 A	1.55 A	1.55 A	1.34 D	1.34 D	3.62 T	3.6 T
	12	3.02 A	3.02 A	3.30 A	3.30 A	4.79 A	4.79 A	4.80 A	3.28 A	3.28 A	3.29 A	2.75 D	2.76 D	6.47 T	6.4 T
	18	4.23 A	4.23 A	4.46 A	4.46 A	6.20 A	6.20 A	6.21 A	4.52 A	4.52 A	4.53 A	3.72 D	3.73 D	8.60 T	8.6 T
	25	5.57 A	5.57 A	5.74 A	5.74 A	7.75 A	7.75 A	7.76 A	5.90 A	5.90 A	5.91 A	4.78 D	4.79 D	11.0 T	11. T
1400	5	2.58 T	2.58 T	2.70 T	2.70 T	3.16 A	3.16 A	3.16 A	2.51 T	2.51 T	2.51 T	1.88 A	1.88 A	3.97 T	3.9 T
	12	5.96 T	5.96 T	5.73 T	5.73 T	5.58 A	5.58 A	5.59 A	5.51 T	5.52 T	5.52 T	4.01 A	4.02 A	7.24 T	7.2 T
	18	8.53 T	8.53 T	8.00 T	8.00 T	7.24 A	7.24 A	7.25 A	7.73 T	7.74 T	7.74 T	5.49 A	5.50 A	9.61 T	9.6 T
	25	11.4 T	11.4 T	10.5 T	10.5 T	9.05 A	9.05 A	9.06 A	10.2 T	10.2 T	10.2 T	7.12 A	7.13 A	12.3 T	12. T
2000	5	2.86 T	2.86 T	2.98 T	2.98 T	4.00 T	4.00 T	4.01 T	2.83 T	2.83 T	2.83 T	2.25 A	2.26 A		
	12	6.59 T	6.59 T	6.36 T	6.36 T	7.61 T	7.61 T	7.61 T	6.19 T	6.19 T	6.20 T	4.79 A	4.81 A		
	18	9.37 T	9.37 T	8.84 T	8.84 T	10.2 T	10.2 T	10.2 T	8.63 T	8.63 T	8.63 T	6.52 A	6.53 A		
	25	12.5 T	12.5 T	11.6 T	11.6 T	13.0 T	13.0 T	13.1 T	11.3 T	11.3 T	11.3 T	8.40 A	8.42 A		
		Payload Power = 2.00 kW													
200	5	1.02 A	1.02 A	1.21 D	1.21 D	2.44 A	2.44 A	2.44 A	0.97 D	0.98 D	0.98 D	0.84 L	0.84 L	3.24 T	3.24 T
	12	2.26 A	2.26 A	2.19 D	2.19 D	4.01 A	4.01 A	4.02 A	1.92 D	1.92 D	1.92 D	1.58 L	1.59 L	5.60 T	5.61 T
	18	3.20 A	3.20 A	2.91 D	2.91 D	5.16 A	5.16 A	5.16 A	2.60 D	2.61 D	2.61 D	2.09 L	2.10 L	7.43 T	7.45 T
	25	4.27 A	4.27 A	3.72 D	3.73 D	6.43 A	6.43 A	6.44 A	3.37 D	3.37 D	3.38 D	2.65 L	2.65 L	9.52 T	9.54 T
800	5	2.32 T	2.32 T	2.44 T	2.44 T	2.88 A	2.88 A	2.89 A	1.58 A	1.59 A	1.59 A	1.60 A	1.61 A	3.65 T	3.66 T
	12	5.34 T	5.34 T	5.11 T	5.11 T	4.94 A	4.94 A	4.95 A	3.33 A	3.34 A	3.34 A	3.37 A	3.38 A	6.51 T	6.53 T
	18	7.70 T	7.70 T	7.17 T	7.17 T	6.38 A	6.38 A	6.39 A	4.59 A	4.59 A	4.59 A	4.64 A	4.64 A	8.65 T	8.67 T
	25	10.4 T	10.4 T	9.50 T	9.50 T	7.97 A	7.97 A	7.98 A	5.97 A	5.98 A	5.98 A	6.04 A	6.04 A	11.1 T	11.1 T
1400	5	2.62 T	2.62 T	2.75 T	2.75 T	3.77 T	3.77 T	3.78 T	2.54 T	2.54 T	2.55 T	1.95 A	1.96 A	3.99 T	4.00 T
	12	6.04 T	6.04 T	5.81 T	5.81 T	7.06 T	7.06 T	7.06 T	5.57 T	5.57 T	5.57 T	4.15 A	4.16 A	7.27 T	7.29 T
	18	8.63 T	8.63 T	8.10 T	8.10 T	9.44 T	9.44 T	9.45 T	7.80 T	7.80 T	7.80 T	5.67 A	5.69 A	9.66 T	9.69 T
	25	11.6 T	11.6 T	10.7 T	10.7 T	12.1 T	12.1 T	12.1 T	10.3 T	10.3 T	10.3 T	7.33 A	7.35 A	12.3 T	12.4 T
2000	5	2.90 T	2.90 T	3.02 T	3.02 T	4.05 T	4.05 T	4.06 T	2.86 T	2.86 T	2.86 T	2.82 T	2.82 T		
	12	6.67 T	6.67 T	6.44 T	6.44 T	7.70 T	7.70 T	7.70 T	6.24 T	6.24 T	6.25 T	6.20 T	6.21 T		
	18	9.47 T	9.48 T	8.94 T	8.94 T	10.3 T	10.3 T	10.3 T	8.69 T	8.69 T	8.70 T	8.65 T	8.66 T		
	25	12.6 T	12.6 T	11.7 T	11.7 T	13.1 T	13.2 T	13.2 T	11.4 T	11.4 T	11.4 T	11.4 T	11.4 T		

[a] T= Titan, A= Atlas, D = Delta, L = LLV3

Technology too heavy for Titan IV

Table E-1. ORM 1 (GEO) (continued)

Payload Mass (kg)	Total # Sat.	Baseline		Advanced Cryo		Nuclear Bimodal			Solar Bimodal			Solar Thermal		Nuclear Electric	
	Lift	Chem	Chem	Cryo	Cryo	H_2	H_2	H_2	H_2	H_2	H_2	H_2	H_2	NH_3 Arc	NH_3 Arc
	Hold	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	N_2H_4 Arc	NH_3 Arc	NH_3 Arc
	Move	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4	Chem	NH_3	N_2H_4	Chem	NH_3	N_2H_4 Arc	Chem	NH_3 Arc	Chem
		Cost ($B FY 95) per Constellation Using Specified Launch Vehicle[a] (MMD/sat. = 10 yr, num. moves/sat. = 3.25)													
		Payload Power = 3.50 kW													
200	5	1.06 A	1.06 A	1.40 A	1.40 A	2.49 A	2.49 A	2.49 A	1.05 D	1.05 D	1.05 D	1.06 D	1.06 D	3.26 T	3.27 T
	12	2.33 A	2.33 A	2.61 A	2.61 A	4.09 A	4.09 A	4.10 A	2.04 D	2.04 D	2.04 D	2.07 D	2.07 D	5.64 T	5.65 T
	18	3.30 A	3.30 A	3.53 A	3.53 A	5.26 A	5.26 A	5.27 A	2.75 D	2.76 D	2.76 D	2.80 D	2.80 D	7.48 T	7.50 T
	25	4.39 A	4.39 A	4.56 A	4.56 A	6.55 A	6.55 A	6.56 A	3.55 D	3.56 D	3.56 D	3.60 D	3.61 D	9.57 T	9.60 T
800	5	2.36 T	2.36 T	2.48 T	2.48 T	2.94 A	2.94 A	2.95 A	1.64 A	1.64 A	1.64 A	1.66 A	1.66 A	3.67 T	3.68 T
	12	5.42 T	5.42 T	5.19 T	5.19 T	5.04 A	5.04 A	5.05 A	3.42 A	3.43 A	3.43 A	3.47 A	3.48 A	6.55 T	6.57 T
	18	7.79 T	7.80 T	7.26 T	7.26 T	6.50 A	6.50 A	6.51 A	4.70 A	4.70 A	4.71 A	4.76 A	4.77 A	8.69 T	8.72 T
	25	10.5 T	10.5 T	9.61 T	9.61 T	8.10 A	8.10 A	8.11 A	6.11 A	6.12 A	6.12 A	6.19 A	6.20 A	11.1 T	11.1 T
1400	5	2.66 T	2.66 T	2.79 T	2.79 T	3.82 T	3.82 T	3.82 T	2.60 T	2.60 T	2.61 T	2.06 A	2.07 A	4.01 T	4.03 T
	12	6.11 T	6.11 T	5.88 T	5.88 T	7.13 T	7.13 T	7.14 T	5.64 T	5.64 T	5.65 T	4.32 A	4.33 A	7.31 T	7.33 T
	18	8.72 T	8.73 T	8.19 T	8.19 T	9.54 T	9.54 T	9.55 T	7.88 T	7.89 T	7.89 T	5.87 A	5.89 A	9.70 T	9.73 T
	25	11.7 T	11.7 T	10.8 T	10.8 T	12.2 T	12.2 T	12.2 T	10.4 T	10.4 T	10.4 T	7.58 A	7.60 A	12.4 T	12.4 T
2000	5	2.94 T	2.94 T	3.06 T	3.06 T	4.11 T	4.11 T	4.12 T	2.89 T	2.89 T	2.89 T	2.90 T	2.91 T		
	12	6.74 T	6.74 T	6.51 T	6.51 T	7.79 T	7.79 T	7.80 T	6.29 T	6.29 T	6.29 T	6.33 T	6.33 T		
	18	9.57 T	9.57 T	9.04 T	9.04 T	10.4 T	10.4 T	10.4 T	8.74 T	8.75 T	8.75 T	8.80 T	8.81 T		
	25	12.3 T	12.3 T	11.9 T	11.9 T	13.3 T	13.3 T	13.3 T	11.5 T	11.5 T	11.5 1	11.5 T	11.6 T		
		Payload Power = 5.00 kW													
200	5	2.03 T	2.03 T	1.44 A	2.15 T	2.59 A	2.59 A	2.60 A	1.31 A	1.41 A	1.31 A	1.31 A	1.32 A	3.28 T	3.29 T
	12	4.64 T	4.64 T	2.68 A	4.41 T	4.23 A	4.23 A	4.24 A	2.64 A	2.65 A	2.65 A	2.67 A	2.68 A	5.67 T	5.68 T
	18	6.76 T	6.76 T	3.62 A	6.22 T	5.42 A	5.42 A	5.43 A	3.65 A	3.65 A	3.66 A	3.69 A	3.70 A	7.52 T	7.54 T
	25	9.18 T	9.18 T	4.67 A	8.30 T	6.74 A	6.74 A	6.75 A	4.78 A	4.79 A	4.79 A	4.84 A	4.84 A	9.62 T	9.65 T
800	5	2.40 T	2.40 T	2.52 T	2.52 T	3.55 T	3.55 T	3.55 T	1.74 A	1.75 A	1.74 A	1.73 A	1.76 A	3.69 T	3.70 T
	12	5.49 T	5.49 T	5.26 T	5.26 T	6.50 T	6.50 T	6.51 T	3.57 A	3.59 A	3.58 A	3.59 A	3.64 A	6.58 T	6.60 T
	18	7.88 T	7.88 T	7.35 T	7.35 T	8.69 T	8.69 T	8.70 T	4.87 A	4.90 A	4.88 A	4.91 A	4.96 A	8.73 T	8.75 T
	25	10.6 T	10.6 T	9.72 T	9.72 T	11.1 T	11.1 T	11.1 T	6.31 A	6.35 A	6.32 A	6.37 A	6.42 A	11.1 T	11.1 T
1400	5	2.70 T	2.70 T	2.82 T	2.82 T	3.87 T	3.87 T	3.87 T	2.63 T	2.63 T	2.63 T	2.62 T	2.66 T	4.03 T	
	12	6.18 T	6.18 T	5.95 T	5.95 T	7.21 T	7.21 T	7.22 T	5.69 T	5.69 T	5.69 T	5.72 T	5.75 T	7.34 T	
	18	8.81 T	8.81 T	8.28 T	8.28 T	9.63 T	9.63 T	9.64 T	7.94 T	7.94 T	7.95 T	7.99 T	8.03 T	9.74 T	
	25	11.8 T	11.8 T	10.9 T	10.9 T	12.3 T	12.3 T	12.3 T	10.5 T	10.5 T	10.5 T	10.5 T	10.6 T	12.4 T	
2000	5	2.97 T	2.97 T	3.10 T	3.10 T	4.16 T	4.16 T	4.28 T	2.94 T	2.95 T	2.94 T	2.95 T	2.96 T		
	12	6.81 T	6.81 T	6.58 T	6.58 T	7.86 T	7.86 T	7.99 T	6.37 T	6.39 T	6.38 T	6.42 T	6.42 T		
	18	9.66 T	9.66 T	9.12 T	9.12 T	10.5 T	10.5 T	10.6 T	8.85 T	8.87 T	8.86 T	8.91 T	8.92 T		
	25	12.8 T	12.8 T	12.0 T	12.0 T	13.4 T	13.4 T	13.6 T	11.6 T	11.6 T	11.6 T	11.7 T	11.7 T		

[a]T = Titan, A = Atlas, D = Delta, L = LLV3

Technology too heavy for Titan IV

Table E-1. ORM 1 (GEO) (continued)

Payload Mass (kg)	Total # Sat.	Nuclear Electric						Solar Electric									
	Lift	H_2 Arc	H_2 Arc	SPT	SPT	Xe Ion	Xe Ion	NH_3 Arc	NH_3 Arc	H_2 Arc	H_2 Arc	SPT	SPT	Xe Ion	Xe Ion
	Hold	N_2H_4 Arc	N_2H_4 Arc	SPT	SPT	Xe Ion	Xe Ion	NH_3 Arc	NH_3 Arc	N_2H_4 Arc	N_2H_4 Arc	SPT	SPT	Xe Ion	Xe Ion
	Move	N_2H_4 Arc	Chem	SPT	Chem	Xe Ion	Chem	NH_3 Arc	Chem	N_2H_4 Arc	Chem	SPT	Chem	Xe Ion	Chem
		Cost ($B FY 95) per Constellation Using Specified Launch Vehicle[a] (MMD/sat. = 10 yr, num. moves/sat. = 3.25)															
		Payload Power = 0.50 kW															
200	5	3.30 T	3.31 T	2.37 A	2.43 A	2.37 A	2.39 A	0.73 L	0.73 L	0.79 L	0.79 L	0.71 L	0.72 L	0.72 L	0.73 L
	12	5.67 T	5.68 T	3.82 A	3.89 A	3.83 A	3.86 A	1.45 L	1.46 L	1.52 L	1.53 L	1.41 L	1.42 L	1.43 L	1.44 L
	18	7.51 T	7.53 T	4.90 A	4.98 A	4.91 A	4.94 A	1.94 L	1.95 L	2.02 L	2.03 L	1.90 L	1.91 L	1.92 L	1.93 L
	25	9.60 T	9.62 T	6.10 A	6.19 A	6.12 A	6.16 A	2.48 L	2.50 L	2.57 L	2.58 L	2.43 L	2.44 L	2.45 L	2.47 L
800	5	3.72 T	3.73 T	3.49 T	3.62 T	2.87 A	2.89 A	1.50 A	1.52 A	1.55 A	1.55 A	1.30 D	1.32 D	1.16 L	1.17 L
	12	6.60 T	6.62 T	6.32 T	6.46 T	4.86 A	4.90 A	3.25 A	3.29 A	3.30 A	3.31 A	2.76 D	2.78 D	2.42 L	2.44 L
	18	8.75 T	8.77 T	8.44 T	8.58 T	6.25 A	6.30 A	4.50 A	4.55 A	4.55 A	4.57 A	3.76 D	3.78 D	3.23 L	3.26 L
	25	11.2 T	11.2 T	10.8 T	11.0 T	7.79 A	7.84 A	5.88 A	5.95 A	5.93 A	5.96 A	4.84 D	4.88 D	4.11 L	4.14 L
1400	5	4.08 T	4.08 T	3.95 T	3.96 T	3.85 T	3.86 T	2.54 T	2.55 T	2.03 A	2.63 T	1.85 A	1.87 A	1.83 A	1.85 A
	12	7.38 T	7.39 T	7.20 T	7.22 T	7.11 T	7.13 T	5.62 T	5.64 T	4.30 A	5.74 T	4.04 A	4.08 A	3.99 A	4.04 A
	18	9.77 T	9.79 T	9.56 T	9.59 T	9.48 T	9.51 T	7.88 T	7.91 T	5.86 A	8.02 T	5.56 A	5.61 A	5.49 A	5.55 A
	25	12.4 T	12.5 T	12.2 T	12.2 T	12.1 T	12.2 T	10.4 T	10.4 T	7.57 A	10.6 T	7.22 A	7.28 A	7.13 A	7.21 A
2000	5	4.40 T	4.41 T	4.26 T	4.27 T	4.16 T	4.29 T	2.91 T	2.92 T	2.96 T	2.97 T	2.85 T	2.87 T	2.23 A	2.26 A
	12	8.08 T	8.10 T	7.88 T	7.91 T	7.80 T	7.94 T	6.40 T	6.42 T	6.47 T	6.49 T	6.32 T	6.35 T	4.87 A	4.93 A
	18	10.7 T	10.7 T	10.5 T	10.5 T	10.4 T	10.6 T	8.91 T	8.94 T	9.00 T	9.02 T	8.82 T	8.86 T	6.65 A	6.73 A
	25	13.6 T	13.6 T	13.3 T	13.4 T	13.3 T	13.4 T	11.7 T	11.7 T	11.8 T	11.8 T	11.6 T	11.6 T	8.59 A	8.69 A
		Payload Power = 2.00 kW															
200	5	3.33 T	3.34 T	2.45 A	2.48 A	2.41 A	2.43 A	0.95 D	1.11 A	1.01 D	1.02 D	0.78 L	0.79 L	0.78 L	0.79 L
	12	5.72 T	5.73 T	3.93 A	3.98 A	3.90 A	3.92 A	1.95 D	2.36 A	2.02 D	2.03 D	1.54 L	1.55 L	1.55 L	1.56 L
	18	7.57 T	7.59 T	5.02 A	5.08 A	4.99 A	5.02 A	2.67 D	3.31 A	2.74 D	2.75 D	2.05 L	2.07 L	2.07 L	2.09 L
	25	9.68 T	9.69 T	6.24 A	6.32 A	6.21 A	6.25 A	3.47 D	4.39 A	3.54 D	3.56 D	2.62 L	2.64 L	2.63 L	2.66 L
800	5	3.75 T	3.76 T	3.63 T	3.64 T	2.91 A	2.93 A	1.64 A	2.24 T	1.64 A	1.65 A	1.52 A	1.53 A	1.38 D	1.39 D
	12	6.65 T	6.66 T	6.48 T	6.50 T	4.91 A	4.96 A	3.49 A	4.93 T	3.47 A	3.49 A	3.27 A	3.29 A	2.90 D	2.92 D
	18	8.80 T	8.82 T	8.61 T	8.63 T	6.32 A	6.37 A	4.80 A	6.96 T	4.77 A	4.80 A	4.52 A	4.55 A	3.93 D	3.96 D
	25	11.2 T	11.2 T	11.0 T	11.0 T	7.86 A	7.93 A	6.25 A	9.25 T	6.21 A	6.24 A	5.91 A	5.95 A	5.05 D	5.10 D
1400	5	4.10 T	4.11 T	3.97 T	3.98 T	3.87 T	3.88 T	2.59 T	2.60 T	2.68 T	2.69 T	1.96 A	1.98 A	1.91 A	1.94 A
	12	7.42 T	7.44 T	7.23 T	7.25 T	7.14 T	7.17 T	5.72 T	5.74 T	5.83 T	5.85 T	4.22 A	4.26 A	4.15 A	4.20 A
	18	9.83 T	9.85 T	9.61 T	9.63 T	9.52 T	9.55 T	8.01 T	8.04 T	8.14 T	8.16 T	5.77 A	5.83 A	5.69 A	5.76 A
	25	12.5 T	12.5 T	12.3 T	12.3 T	12.1 T	12.2 T	10.6 T	10.6 T	10.7 T	10.7 T	7.47 A	7.54 A	7.38 A	7.46 A
2000	5	4.42 T	4.43 T	4.28 T	4.29 T	4.30 T	4.32 T	2.96 T	2.98 T	3.02 T	3.03 T	2.90 T	2.92 T	2.30 A	2.36 A
	12	8.12 T	8.14 T	7.92 T	7.94 T	7.95 T	7.98 T	6.50 T	6.52 T	6.58 T	6.60 T	6.42 T	6.45 T	5.00 A	5.09 A
	18	10.8 T	10.8 T	10.5 T	10.5 T	10.6 T	10.6 T	9.04 T	9.07 T	9.14 T	9.16 T	8.94 T	8.98 T	6.82 A	6.93 A
	25	13.7 T	13.7 T	13.4 T	13.4 T	13.4 T	13.5 T	11.8 T	11.8 T	11.9 T	11.9 T	11.7 T	11.7 T	8.80 A	8.93 A

[a]T = Titan, A = Atlas, D = Delta, L = LLV3

Table E-1. ORM 1 (GEO) (concluded)

		Nuclear Electric						Solar Electric							
	Lift	H_2 Arc	H_2 Arc	SPT	SPT	Xe Ion	Xe Ion	NH_3 Arc	NH_3 Arc	H_2 Arc	H_2 Arc	SPT	SPT	Xe Ion	Xe Ion
	Hold	N_2H_4 Arc	N_2H_4 Arc	SPT	SPT	Xe Ion	Xe Ion	NH_3 Arc	NH_3 Arc	N_2H_4 Arc	N_2H_4 Arc	SPT	SPT	Xe Ion	Xe Ion
	Move	N_2H_4 Arc	Chem	SPT	Chem	Xe Ion	Chem	NH_3 Arc	Chem	N_2H_4 Arc	Chem	SPT	Chem	Xe Ion	Chem
Payload Mass (kg)	Total # Sat.	Cost ($B FY 95) per Constellation Using Specified Launch Vehicle[a] (MMD/sat. = 10 yr, num. moves/sat. = 3.25)													
Payload Power = 3.50 kW															
200	5	3.36 T	3.36 T	2.50 A	2.53 A	2.45 A	2.50 A	1.17 A	1.18 A	1.22 A	1.22 A	0.99 D	0.99 D	0.84 L	0.85 L
	12	5.76 T	5.77 T	4.00 A	4.05 A	3.94 A	4.01 A	2.46 A	2.48 A	2.52 A	2.52 A	2.00 D	2.02 D	1.65 L	1.67 L
	18	7.62 T	7.64 T	5.11 A	5.17 A	5.05 A	5.12 A	3.44 A	3.46 A	3.50 A	3.50 A	2.73 D	2.75 D	2.20 L	2.23 L
	25	9.73 T	9.75 T	6.35 A	6.42 A	6.28 A	6.36 A	4.54 A	4.57 A	4.61 A	4.61 A	3.54 D	3.57 D	2.80 L	2.83 L
800	5	3.78 T	3.78 T	3.65 T	3.66 T	2.94 A	2.97 A	2.28 T	2.29 T	1.76 A	1.77 A	1.59 A	1.61 A	1.44 D	1.58 A
	12	6.69 T	6.70 T	6.51 T	6.53 T	4.96 A	5.01 A	5.01 T	5.02 T	3.66 A	3.68 A	3.41 A	3.44 A	3.01 D	3.39 A
	18	8.85 T	8.87 T	8.65 T	8.67 T	6.38 A	6.44 A	7.06 T	7.08 T	5.00 A	5.03 A	4.70 A	4.75 A	4.07 D	4.68 A
	25	11.3 T	11.3 T	11.0 T	11.1 T	7.93 A	8.01 A	9.37 T	9.40 T	6.47 A	6.52 A	6.13 A	6.19 A	5.23 D	6.11 A
1400	5	4.13 T	4.14 T	3.99 T	4.00 T	3.89 T	4.03 T	2.68 T	2.69 T	2.74 T	2.74 T	2.03 A	2.64 T	1.99 A	2.01 A
	12	7.46 T	7.47 T	7.27 T	7.29 T	7.18 T	7.32 T	5.85 T	5.87 T	5.93 T	5.94 T	4.35 A	5.80 T	4.29 A	4.34 A
	18	9.87 T	9.89 T	9.65 T	9.67 T	9.56 T	9.71 T	8.17 T	8.20 T	8.26 T	8.28 T	5.94 A	8.12 T	5.87 A	5.94 A
	25	12.6 T	12.6 T	12.3 T	12.3 T	12.2 T	12.4 T	10.8 T	10.8 T	10.9 T	10.9 T	7.68 A	10.7 T	7.60 A	7.69 A
2000	5	4.45 T	4.46 T	4.30 T	4.32 T	4.32 T	4.34 T	3.01 T	3.03 T	3.11 T	3.12 T	2.95 T	2.97 T	2.40 A	3.01 T
	12	8.16 T	8.18 T	7.95 T	7.97 T	7.98 T	8.01 T	6.59 T	6.62 T	6.72 T	6.74 T	6.51 T	6.54 T	5.16 A	6.63 T
	18	10.8 T	10.8 T	10.6 T	10.6 T	10.6 T	10.6 T	9.15 T	9.19 T	9.30 T	9.32 T	9.06 T	9.10 T	7.02 A	9.21 T
	25	13.7 T	13.8 T	13.4 T	13.5 T	13.5 T	13.5 T	12.0 T	12.0 T	12.2 T	12.2 T	11.9 T	11.9 T	9.04 A	12.1 T
Payload Power = 5.00 kW															
200	5	3.38 T	3.39 T	2.54 A	2.69 A	2.51 A	2.53 A	1.28 A	1.30 A	1.29 A	1.30 A	1.04 D	1.05 D	0.90 L	1.06 D
	12	5.79 T	5.81 T	4.07 A	4.23 A	4.03 A	4.06 A	2.65 A	2.68 A	2.65 A	2.67 A	2.10 D	2.12 D	1.75 L	2.14 D
	18	7.66 T	7.68 T	5.18 A	5.36 A	5.14 A	5.18 A	3.68 A	3.72 A	3.67 A	3.70 A	2.86 D	2.89 D	2.33 L	2.90 D
	25	9.78 T	9.80 T	6.43 A	6.62 A	6.38 A	6.42 A	4.83 A	4.88 A	4.81 A	4.85 A	3.70 D	3.73 D	2.96 L	3.75 D
800	5	3.80 T	3.81 T	3.67 T	3.68 T	2.97 A	2.99 A	2.33 T	2.34 T	2.42 T	2.43 T	1.66 A	1.71 A	1.63 A	1.66 A
	12	6.72 T	6.73 T	6.54 T	6.56 T	5.01 A	5.05 A	5.09 T	5.11 T	5.21 T	5.22 T	3.53 A	3.60 A	3.49 A	3.53 A
	18	8.89 T	8.91 T	8.68 T	8.71 T	6.44 A	6.49 A	7.17 T	7.19 T	7.30 T	7.31 T	4.86 A	4.94 A	4.81 A	4.86 A
	25	11.3 T	11.3 T	11.0 T	11.1 T	8.00 A	8.07 A	9.51 T	9.54 T	9.65 T	9.67 T	6.33 A	6.42 A	6.26 A	6.33 A
1400	5	4.15 T	4.16 T	4.01 T	4.02 T	3.91 T	4.05 T	2.73 T	2.74 T	2.79 T	2.79 T	2.67 T	2.69 T	2.08 A	2.11 A
	12	7.49 T	7.51 T	7.29 T	7.32 T	7.21 T	7.35 T	5.94 T	5.96 T	6.02 T	6.04 T	5.86 T	5.89 T	4.45 A	4.50 A
	18	9.91 T	9.93 T	9.68 T	9.71 T	9.60 T	9.75 T	8.28 T	8.31 T	8.38 T	8.40 T	8.19 T	8.23 T	6.07 A	6.14 A
	25	12.6 T	12.6 T	12.3 T	12.4 T	12.3 T	12.4 T	10.9 T	11.0 T	11.0 T	11.0 T	10.8 T	10.8 T	7.83 A	7.93 A
2000	5	4.47 T	4.48 T	4.32 T	4.33 T	4.34 T	4.36 T	3.06 T	3.08 T	3.16 T	3.17 T	3.00 T	3.02 T	3.04 T	3.06 T
	12	8.19 T	8.21 T	7.98 T	8.00 T	8.01 T	8.04 T	6.68 T	6.72 T	6.81 T	6.83 T	6.59 T	6.63 T	6.68 T	6.72 T
	18	10.8 T	10.9 T	10.6 T	10.6 T	10.6 T	10.7 T	9.27 T	9.31 T	9.41 T	9.44 T	9.17 T	9.21 T	9.28 T	9.34 T

[a]T = Titan, A = Atlas, D = Delta, L = LLV3

Table E-2. ORM 2a (GPS): Estimated Cost of Establishing and Maintaining Constellations for 15 Years

Payload Mass (kg)	Total # Sat.	Baseline	Adv. Cryo	Nuclear Bimodal	Solar Bimodal	Solar Thermal	Nuclear Electric				Solar Electric			
	Lift	Chem	Cryo	H_2	H_2	H_2	NH_3 Arc	H_2 Arc	SPT	Xe Ion	NH_3 Arc	H_2 Arc	SPT	Xe Ion
	Hold	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem
	Move	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem
		Cost ($B FY 95) per Constellation Using Specified Launch Vehicle[a] (MMD/sat. = 10 yr, num. moves/sat. = 3)												
		Payload Power = 0.5 kW												
200	45	4.72 D	5.51 D	9.44 A	3.59 L	3.50 L	8.72 A	7.25 D	7.12 D	7.23 D	3.44 L	3.55 L	3.43 L	3.48 L
	53	5.47 D	6.33 D	10.8 A	4.10 L	4.00 L	9.95 A	8.16 D	8.02 D	8.14 D	3.94 L	4.06 L	3.93 L	3.99 L
	62	6.30 D	7.24 D	12.2 A	4.67 L	4.56 L	11.3 A	9.18 D	9.03 D	9.16 D	4.50 L	4.62 L	4.49 L	4.55 L
	70	7.04 D	8.05 D	13.5 A	5.17 L	5.05 L	12.5 A	10.1 D	9.91 D	10.1 D	4.99 L	5.12 L	4.98 L	5.05 L
300	45	5.03 D	5.82 D	9.75 A	3.93 L	3.87 L	9.04 A	9.07 A	9.03 A	7.60 D	3.82 L	3.93 L	3.77 L	3.84 L
	53	5.83 D	6.68 D	11.1 A	4.48 L	4.42 L	10.3 A	10.3 A	10.3 A	8.56 D	4.37 L	4.49 L	4.32 L	4.39 L
	62	6.71 D	7.64 D	12.6 A	5.10 L	5.03 L	11.7 A	11.8 A	11.7 A	9.63 D	4.99 L	5.11 L	4.92 L	5.00 L
	70	7.48 D	8.49 D	13.9 A	5.64 L	5.57 L	13.0 A	13.0 A	13.0 A	10.6 D	5.52 L	5.66 L	5.46 L	5.54 L
400	45	5.36 D	6.15 D	10.0 A	4.24 L	4.18 L	9.33 A	9.37 A	9.32 A	9.35 A	4.16 L	4.27 L	4.15 L	4.21 L
	53	6.19 D	7.05 D	11.4 A	4.83 L	4.77 L	10.6 A	10.7 A	10.6 A	10.7 A	4.75 L	4.87 L	4.74 L	4.81 L
	62	7.12 D	8.05 D	13.0 A	5.49 L	5.42 L	12.1 A	12.1 A	12.1 A	12.1 A	5.41 L	5.53 L	5.40 L	5.47 L
	70	7.93 D	8.94 D	14.3 A	6.07 L	6.00 L	13.4 A	13.4 A	13.4 A	13.4 A	5.99 L	6.12 L	5.98 L	6.05 L
500	45	7.70 A	6.42 D	10.3 A	4.53 L	4.48 L	9.6 A	9.69 A	9.60 A	9.67 A	4.46 L	4.57 L	4.46 L	4.52 L
	53	8.94 A	7.36 D	11.7 A	5.16 L	5.10 L	10.9 A	11.0 A	10.9 A	11.0 A	5.10 L	5.22 L	5.09 L	5.16 L
	62	10.3 A	8.40 D	13.3 A	5.86 L	5.80 L	12.4 A	12.5 A	12.4 A	12.5 A	5.80 L	5.92 L	5.79 L	5.87 L
	70	11.5 A	9.32 D	14.7 A	6.48 L	6.41 L	13.8 A	13.9 A	13.8 A	13.8 A	6.41 L	6.55 L	6.40 L	6.49 L
		Payload Power = 0.83 kW												
200	45	4.77 D	5.56 D	9.48 A	3.62 L	3.56 L	8.77 A	8.80 A	7.16 D	7.25 D	3.53 L	3.64 L	3.48 L	3.54 L
	53	5.52 D	6.38 D	10.8 A	4.14 L	4.07 L	10.0 A	10.0 A	8.06 D	8.17 D	4.04 L	4.16 L	3.99 L	4.05 L
	62	6.36 D	7.30 D	12.3 A	4.71 L	4.64 L	11.4 A	11.4 A	9.07 D	9.19 D	4.61 L	4.74 L	4.55 L	4.62 L
	70	7.11 D	8.12 D	13.6 A	5.21 L	5.13 L	12.6 A	12.9 A	9.96 D	10.1 D	5.11 L	5.24 L	5.04 L	5.12 L
300	45	5.09 D	5.87 D	9.80 A	3.96 L	3.94 L	9.08 A	9.12 A	9.06 A	7.63 D	3.88 L	4.00 L	3.88 L	3.89 L
	53	5.88 D	6.74 D	11.2 A	4.52 L	4.49 L	10.4 A	10.4 A	10.3 A	8.59 D	4.43 L	4.56 L	4.44 L	4.45 L
	62	6.76 D	7.70 D	12.7 A	5.14 L	5.11 L	11.8 A	11.8 A	11.8 A	9.66 D	5.05 L	5.19 L	5.06 L	5.07 L
	70	7.55 D	8.56 D	14.0 A	5.68 L	5.65 L	13.0 A	13.1 A	13.0 A	10.6 D	5.59 L	5.74 L	5.60 L	5.62 L
400	45	5.41 A	6.19 D	10.1 A	4.27 L	4.25 L	9.40 A	9.44 A	9.35 A	9.41 A	4.21 L	4.32 L	4.20 L	4.27 L
	53	6.24 A	7.10 D	11.5 A	4.87 L	4.84 L	10.7 A	10.8 A	10.7 A	10.7 A	4.81 L	4.93 L	4.80 L	4.87 L
	62	7.17 A	8.11 D	13.0 A	5.53 L	5.50 L	12.2 A	12.2 A	12.1 A	12.2 A	5.48 L	5.60 L	5.47 L	5.54 L
	70	7.99 A	9.01 D	14.4 A	6.11 L	6.08 L	13.5 A	13.5 A	13.4 A	13.5 A	6.06 L	6.19 L	6.05 L	6.13 L
500	45	7.75 A	6.47 D	10.4 A	4.56 L	4.54 L	9.67 A	9.72 A	9.66 A	9.69 A	4.52 L	4.63 L	4.51 L	4.59 L
	53	8.99 A	7.41 D	11.8 A	5.20 L	5.18 L	11.0 A	11.1 A	11.0 A	11.0 A	5.16 L	5.28 L	5.15 L	5.23 L
	62	10.4 A	8.46 D	13.4 A	5.91 L	5.88 L	12.5 A	12.6 A	12.5 A	12.5 A	4.87 L	5.99 L	5.85 L	5.95 L
	70	11.6 A	9.38 D	14.8 A	6.52 L	6.49 L	13.9 A	13.9 A	13.8 A	13.9 A	6.49 L	6.62 L	6.47 L	6.58 L

[a]T = Titan, A = Atlas, D = Delta, L = LLV3

Table E-2. ORM 2a (GPS) (concluded)

		Baseline	Adv. Cryo	Nuclear Bimodal	Solar Bimodal	Solar Thermal	Nuclear Electric				Solar Electric			
	Lift	Chem	Cryo	H_2	H_2	H_2	NH_3 Arc	H_2 Arc	SPT	Xe Ion	NH_3 Arc	H_2 Arc	SPT	Xe Ion
	Hold	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem
	Move	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem	Chem
Payload Mass (kg)	Total # Sat.	Cost ($B FY 95) per Constellation Using Specified Launch Vehicle[a] (MMD/sat. = 10 yr, num. moves/sat. = 3)												
Payload Power = 1.17 kW														
200	45	4.82 D	5.60 D	9.53 A	3.65 L	3.66 L	8.80 A	8.84 A	7.19 D	7.28 D	3.58 L	3.70 L	3.55 L	3.59 L
	53	5.58 D	6.43 D	10.9 A	4.17 L	4.18 L	10.0 A	10.1 A	8.10 D	8.21 D	4.10 L	4.23 L	4.06 L	4.11 L
	62	6.42 D	7.36 D	12.3 A	4.75 L	4.76 L	11.4 A	11.5 A	9.11 D	9.23 D	4.67 L	4.81 L	4.63 L	4.69 L
	70	7.17 D	8.18 D	13.6 A	5.25 L	5.27 L	12.6 A	12.7 A	10.0 D	10.1 D	5.18 L	5.32 L	5.13 L	5.20 L
300	45	5.17 D	5.95 D	9.84 A	3.99 L	4.00 L	9.15 A	9.19 A	9.10 A	7.65 D	3.94 L	4.05 L	3.93 L	3.99 L
	53	5.97 D	6.83 D	11.2 A	4.55 L	4.56 L	10.4 A	10.5 A	10.4 A	8.62 D	4.50 L	4.62 L	4.50 L	4.56 L
	62	6.87 D	7.80 D	12.7 A	5.17 L	5.19 L	11.9 A	11.9 A	11.8 A	9.69 D	5.13 L	5.25 L	5.12 L	5.19 L
	70	7.65 D	8.67 D	14.1 A	5.72 L	5.74 L	13.1 A	13.2 A	13.1 A	10.6 D	5.68 L	5.81 L	5.67 L	5.75 L
400	45	7.53 A	6.24 D	10.1 A	4.30 L	4.31 L	9.44 A	9.48 A	9.42 A	9.42 A	4.26 L	4.38 L	4.26 L	4.32 L
	53	8.73 A	7.15 D	11.5 A	4.90 L	4.92 L	10.8 A	10.8 A	10.7 A	10.4 A	4.87 L	4.99 L	4.86 L	4.94 L
	62	10.1 A	8.17 D	13.1 A	5.57 L	5.58 L	12.2 A	12.3 A	12.2 A	12.2 A	5.54 L	5.67 L	5.53 L	5.62 L
	70	11.3 A	9.07 D	14.5 A	6.15 L	6.17 L	13.5 A	13.6 A	13.5 A	13.5 A	6.13 L	6.27 L	6.12 L	6.21 L
500	45	7.80 A	6.51 D	10.4 A	4.60 L	4.61 L	9.71 A	9.75 A	9.69 A	9.70 A	4.58 L	4.69 L	4.56 L	4.65 L
	53	9.04 A	7.46 D	11.8 A	5.24 L	5.25 L	11.1 A	11.1 A	11.0 A	11.1 A	5.22 L	5.34 L	5.21 L	5.30 L
	62	10.4 A	8.52 D	13.4 A	5.95 L	5.96 L	12.6 A	12.6 A	12.6 A	12.6 A	5.94 L	6.06 L	5.92 L	6.02 L
	70	11.7 A	9.45 D	14.8 A	6.57 L	6.58 L	13.9 A	13.9 A	13.9 A	13.9 A	6.57 L	6.70 L	6.55 L	6.66 L
Payload Power = 1.5 kW														
200	45	4.89 D	5.68 D	9.57 A	3.68 L	3.72 L	8.87 A	8.91 A	8.82 A	7.31 D	3.64 L	3.75 L	3.63 L	3.65 L
	53	5.66 D	6.52 D	10.9 A	4.20 L	4.25 L	10.1 A	10.2 A	10.1 A	8.24 D	4.16 L	4.28 L	4.16 L	4.17 L
	62	6.52 D	7.46 D	12.4 A	4.78 L	4.83 L	11.5 A	11.4 A	11.4 A	9.26 D	4.75 L	4.87 L	4.74 L	4.76 L
	70	7.28 D	8.29 D	13.7 A	5.29 L	5.35 L	12.7 A	12.8 A	12.7 A	10.2 D	5.26 L	5.39 L	5.25 L	5.27 L
300	45	7.28 A	5.99 D	9.88 A	4.02 L	4.06 L	9.19 A	9.23 A	9.17 A	9.17 A	3.99 L	4.10 L	3.98 L	4.04 L
	53	8.46 A	6.88 D	11.3 A	4.58 L	4.63 L	10.5 A	10.5 A	10.5 A	10.5 A	4.56 L	4.68 L	4.55 L	4.62 L
	62	9.77 A	7.86 D	12.8 A	5.21 L	5.26 L	11.9 A	11.9 A	11.9 A	11.9 A	5.19 L	5.32 L	5.18 L	5.26 L
	70	10.9 A	8.73 D	14.1 A	5.76 L	5.82 L	13.2 A	13.2 A	13.1 A	13.1 A	5.75 L	5.88 L	5.73 L	5.82 L
400	45	7.57 A	6.28 D	10.2 A	4.33 L	4.37 L	9.47 A	9.52 A	9.46 A	9.46 A	4.32 L	4.43 L	4.31 L	4.38 L
	53	8.78 A	7.20 D	11.6 A	4.93 L	4.98 L	10.8 A	10.8 A	10.8 A	10.8 A	4.93 L	5.05 L	4.92 L	5.00 L
	62	10.1 A	8.23 D	13.1 A	5.60 L	5.66 L	12.3 A	12.3 A	12.2 A	12.2 A	5.61 L	5.73 L	5.59 L	5.69 L
	70	11.3 A	9.13 D	14.5 A	6.19 L	6.25 L	13.6 A	13.6 A	13.5 A	13.5 A	6.20 L	6.33 L	6.19 L	6.29 L
500	45	7.84 A	6.55 D	10.5 A	4.64 L	4.67 L	9.74 A	9.79 A	9.73 A	9.73 A	4.63 L	4.74 L	4.61 L	4.70 L
	53	9.09 A	7.51 D	11.9 A	5.28 L	5.32 L	11.1 A	11.1 A	11.1 A	11.1 A	5.29 L	5.40 L	5.26 L	5.36 L
	62	10.5 A	8.57 D	13.5 A	5.99 L	6.03 L	12.6 A	12.7 A	12.6 A	12.6 A	6.01 L	6.13 L	5.98 L	6.09 L
	70	11.7 A	9.51 D	14.9 A	6.62 L	6.66 L	13.9 A	14.0 A	13.9 A	13.9 A	6.64 L	6.77 L	6.62 L	6.74 L

[a]T = Titan, A = Atlas, D = Delta, L = LLV3

Table E-3. ORM 3a (LEO-Polar): Estimated Cost of Establishing and Maintaining Constellations for 15 Years

		Baseline		Direct Insertion[a]		Advanced Cryo		Solar Electric									
	Lift / Hold / Move	Biprop Chem / Chem / Chem		Direct Insert. / Chem / Chem		Adv. Cryo / Chem / Chem		H_2 Arc / Chem / Chem		NH_3 Arc / Chem / Chem		N_2H_4 Arc / Chem / Chem		Xe SPT / Chem / Chem		Xe Ion / Chem / Chem	
Payload Mass (kg)	Total # Sat.	Cost ($B FY 95) per Constellation Using Specified Launch Vehicle[b] (MMD/sat. = 5.5 yr, num. moves/sat. = 1)															
Payload Power = 0.5 kW																	
500	2	0.41	L	0.40	L	0.87	D	0.51	L	0.49	L	0.48	L	0.49	L	0.49	L
	9	1.50	L	1.49	L	2.25	D	1.61	L	1.58	L	1.57	L	1.58	L	1.59	L
	15	2.27	L	2.25	L	3.26	D	2.39	L	2.35	L	2.33	L	2.35	L	2.36	L
1000	2	0.55	L	0.54	L	1.00	D	0.66	L	0.63	L	0.62	L	0.63	L	0.63	L
	9	2.11	L	2.09	L	2.85	D	2.23	L	2.19	L	2.17	L	2.19	L	2.20	L
	15	3.18	L	3.16	L	4.17	D	3.30	L	3.26	L	3.24	L	3.27	L	3.28	L
1500	2	0.66	L	0.72	D	1.12	D	0.84	D	0.75	L	0.73	L	0.75	L	0.75	L
	9	2.63	L	2.90	D	3.37	D	3.04	D	2.71	L	2.69	L	2.72	L	2.73	L
	15	3.96	L	4.42	D	4.95	D	4.57	D	4.05	L	4.02	L	4.06	L	4.07	L
Payload Power = 1.0 kW																	
500	2	0.42	L	0.42	L	0.88	D	0.53	L	0.50	L	0.49	L	0.50	L	0.50	L
	9	1.54	L	1.52	L	2.28	D	1.65	L	1.62	L	1.60	L	1.62	L	1.62	L
	15	2.31	L	2.30	L	3.31	D	2.43	L	2.40	L	2.38	L	2.40	L	2.40	L
1000	2	0.56	L	0.55	L	1.02	D	0.67	L	0.64	L	0.63	L	0.64	L	0.64	L
	9	2.14	L	2.13	L	2.89	D	2.26	L	2.22	L	2.21	L	2.22	L	2.23	L
	15	3.22	L	3.20	L	4.21	D	3.35	L	3.31	L	3.29	L	3.31	L	3.32	L
1500	2	0.74	D	0.74	D	1.14	D	0.86	D	0.82	D	0.81	D	0.83	D	0.83	D
	9	2.95	D	2.93	D	3.40	D	3.07	D	3.03	D	3.01	D	3.03	D	3.05	D
	15	4.48	D	4.46	D	4.99	D	4.61	D	4.57	D	4.55	D	4.57	D	4.59	D

[a]Direct insertion by the launch vehicle; no upper stage is used.
[b]T = Titan, A = Atlas, D = Delta, L = LLV3

Table E-3. ORM 3a (LEO-Polar) (concluded)

Payload Mass (kg)	Lift / Hold / Move — Total # Sat.	Baseline: Biprop Chem / Chem / Chem		Direct Insertion[a]: Direct Insert. / Chem / Chem		Advanced Cryo: Adv. Cryo / Chem / Chem		Solar Electric: H_2 Arc / Chem / Chem		Solar Electric: NH_3 Arc / Chem / Chem		Solar Electric: N_2H_4 Arc / Chem / Chem		Solar Electric: Xe SPT / Chem / Chem		Solar Electric: Xe Ion / Chem / Chem	
		Cost ($B FY 95) per Constellation Using Specified Launch Vehicle[b] (MMD/sat. = 5.5 yr, num. moves/sat. = 1)															
		Payload Power = 1.5 kW															
500	2	0.44	L	0.43	L	0.90	D	0.55	L	0.52	L	0.51	L	0.52	L	0.52	L
	9	1.58	L	1.55	L	2.32	D	1.70	L	1.66	L	1.65	L	1.66	L	1.66	L
	15	2.37	L	2.34	L	3.36	D	2.50	L	2.46	L	2.44	L	2.46	L	2.46	L
1000	2	0.57	L	0.57	L	1.03	D	0.69	L	0.66	L	0.65	L	0.66	L	0.66	L
	9	2.17	L	2.16	L	2.92	D	2.29	L	2.26	L	2.24	L	2.26	L	2.26	L
	15	3.26	L	3.25	L	4.26	D	3.39	L	3.35	L	3.34	L	3.35	L	3.35	L
1500	2	0.75	D	0.75	D	1.15	D	0.87	D	0.84	D	0.83	D	0.84	D	0.84	D
	9	2.98	D	2.96	D	3.43	D	3.10	D	3.06	D	3.05	D	3.06	D	3.07	D
	15	4.52	D	4.51	D	5.04	D	4.66	D	4.61	D	4.60	D	4.61	D	4.62	D
		Payload Power = 2.0 kW															
500	2	0.45	L	0.44	L	0.91	D	0.56	L	0.53	L	0.52	L	0.53	L	0.53	L
	9	1.61	L	1.59	L	2.35	D	1.73	L	1.70	L	1.68	L	1.70	L	1.69	L
	15	2.41	L	2.40	L	3.41	D	2.54	L	2.51	L	2.49	L	2.51	L	2.50	L
1000	2	0.58	L	0.64	D	1.04	D	0.70	L	0.67	L	0.66	L	0.67	L	0.67	L
	9	2.20	L	2.47	D	2.95	D	2.33	L	2.29	L	2.27	L	2.29	L	2.29	L
	15	3.31	L	3.77	D	4.30	D	3.44	L	3.40	L	3.38	L	3.39	L	3.40	L
1500	2	0.77	D	0.76	D	1.16	D	0.88	D	0.85	D	0.84	D	0.85	D	0.85	D
	9	3.01	D	2.99	D	3.46	D	3.13	D	3.10	D	3.08	D	3.09	D	3.10	D
	15	4.57	D	4.55	D	5.08	D	4.70	D	4.66	D	4.64	D	4.66	D	4.66	D

[a] Direct insertion by the launch vehicle; no upper stage is used.
[b] T = Titan, A = Atlas, D = Delta, L = LLV3

Table E-4. ORM 4 (HEO): Estimated Cost of Establishing and Maintaining Constellations for 15 Years

		Baseline	Direct Insertion[a]	Advanced Cryo		Nuclear Bimodal		Solar Bimodal		Solar Thermal	
	Lift	Direct Insert.	Direct Insert.	Adv. Cryo	Adv. Cryo	H_2	H_2	H_2	H_2	H_2	H_2
	Hold	N_2H_4 Arc	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc
	Move	N_2H_4 Arc	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc
Payload Mass (kg)	Total # Sat.	Cost ($B FY 95) per Constellation Using Specified Launch Vehicle[b] (MMD/sat. = 7.5 yr, num. moves/sat. = 2)									
		Payload Power = 0.5 kW									
500	4	.98 A	0.98 A	1.21 D	1.21 D	2.35 A	2.35 A	0.98 D	0.99 D	0.86 L	0.87 L
	6	1.42 A	1.43 A	1.57 D	1.58 D	2.90 A	2.90 A	1.35 D	1.35 D	1.17 L	1.17 L
	8	1.86 A	1.87 A	1.94 D	1.95 D	3.45 A	3.45 A	1.72 D	1.72 D	1.47 L	1.48 L
1000	4	1.24 A	1.24 A	1.46 D	1.47 D	2.62 A	2.62 A	1.43 A	1.28 D	1.27 D	1.27 D
	6	1.81 A	1.81 A	1.96 D	1.97 D	3.30 A	3.30 A	2.01 A	1.78 D	1.77 D	1.77 D
	8	2.38 A	2.38 A	2.46 D	2.46 D	3.97 A	3.97 A	2.60 A	2.28 D	2.27 D	2.27 D
1500	4	1.96 T	1.95 T	1.81 A	1.82 A	2.90 A	2.90 A	1.70 A	1.70 A	1.69 A	1.69 A
	6	2.88 T	2.88 T	2.47 A	2.48 A	3.69 A	3.69 A	2.41 A	2.40 A	2.40 A	2.40 A
	8	3.81 T	3.80 T	3.14 A	3.14 A	4.48 A	4.48 A	3.12 A	3.11 A	3.10 A	3.10 A
2000	4	2.17 T	2.16 T	2.17 T	2.16 T	3.59 T	3.59 T	2.41 T	1.98 A	1.94 A	1.94 A
	6	3.20 T	3.19 T	3.20 T	3.19 T	4.72 T	4.72 T	3.46 T	2.79 A	2.75 A	2.75 A
	8	4.22 T	4.22 T	4.22 T	4.22 T	5.85 T	5.85 T	4.52 T	3.61 A	3.57 A	3.57 A
		Payload Power = 2.0 kW									
500	4	1.02 A	1.03 A	1.25 D	1.26 D	2.40 A	2.40 A	1.03 D	1.03 D	1.04 D	1.05 D
	6	1.48 A	1.48 A	1.63 D	1.64 D	2.97 A	2.97 A	1.40 D	1.41 D	1.43 D	1.43 D
	8	1.93 A	1.94 A	2.01 D	2.02 D	3.53 A	3.53 A	1.78 D	1.78 D	1.81 D	1.81 D
1000	4	1.77 T	1.28 A	1.62 A	1.63 A	2.72 A	2.72 A	1.47 A	1.47 A	1.49 A	1.49 A
	6	2.59 T	1.86 A	2.18 A	2.19 A	3.40 A	3.40 A	2.06 A	2.06 A	2.09 A	2.09 A
	8	3.42 T	2.44 A	2.75 A	2.75 A	4.09 A	4.09 A	2.66 A	2.66 A	2.70 A	2.70 A
1500	4	1.99 T	1.99 T	1.99 T	1.99 T	2.95 A	2.95 A	1.77 A	1.74 A	1.75 A	1.75 A
	6	2.93 T	2.92 T	2.93 T	2.92 T	3.75 A	3.75 A	2.49 A	2.45 A	2.47 A	2.47 A
	8	3.87 T	3.86 T	3.87 T	3.86 T	4.55 A	4.55 A	3.21 A	3.16 A	3.20 A	3.19 A
2000	4					3.63 T	3.63 T	2.47 T	2.44 T	2.03 A	1.99 A
	6					4.77 T	4.77 T	3.54 T	3.50 T	2.87 A	2.82 A
	8					5.92 T	5.92 T	4.60 T	4.56 T	3.70 A	3.65 A

[a] Direct insertion by the launch vehicle; no upper stage is used.
[b] T = Titan, A = Atlas, D = Delta, L = LLV3

Technology too heavy for Titan IV

Table E-4. ORM 4 (HEO) (concluded)

		Baseline	Direct Insertion[a]	Advanced Cryo		Nuclear Bimodal		Solar Bimodal		Solar Thermal	
	Lift	Direct Insert.	Direct Insert.	Adv. Cryo	Adv. Cryo	H_2	H_2	H_2	H_2	H_2	H_2
	Hold	N_2H_4 Arc	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc
	Move	N_2H_4 Arc	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc	Chem	N_2H_4 Arc
Payload Mass (kg)	Total # Sat.	Cost ($B FY 95) per Constellation Using Specified Launch Vehicle[b] (MMD/sat. = 7.5 yr, num. moves/sat. = 2)									
		Payload Power = 3.0 kW									
500	4	1.05 A	1.06 A	1.40 A	1.29 D	2.45 A	2.45 A	1.08 D	1.09 D	1.10 D	1.10 D
	6	1.52 A	1.53 A	1.85 A	1.68 D	3.02 A	3.02 A	1.47 D	1.48 D	1.50 D	1.50 D
	8	1.99 A	1.99 A	2.30 A	2.07 D	3.59 A	3.59 A	1.86 D	1.87 D	1.90 D	1.90 D
1000	4	1.80 T	1.80 T	1.80 T	1.66 A	2.76 A	2.76 A	1.52 A	1.52 A	1.54 A	1.54 A
	6	2.64 T	2.63 T	2.64 T	2.23 A	3.46 A	3.46 A	2.13 A	2.13 A	2.16 A	2.16 A
	8	3.47 T	3.46 T	3.47 T	2.81 A	4.15 A	4.15 A	2.74 A	2.73 A	2.78 A	2.77 A
1500	4	2.03 T	2.02 T	2.03 T	2.02 T	3.00 A	3.00 A	1.81 A	1.80 A	1.84 A	1.80 A
	6	2.97 T	2.97 T	2.97 T	2.97 T	3.81 A	3.81 A	2.53 A	2.52 A	2.57 A	2.54 A
	8	3.92 T	3.91 T	3.92 T	3.91 T	4.62 A	4.62 A	3.26 A	3.24 A	3.31 A	3.27 A
2000	4					3.67 T	3.67 T	2.50 T	2.50 T	2.48 T	2.48 T
	6					4.83 T	4.83 T	3.57 T	3.57 T	3.55 T	3.55 T
	8					5.98 T	5.98 T	4.64 T	4.63 T	4.63 T	4.62 T
		Payload Power = 5.0 kW									
500	4	1.58 T	1.09 A	1.43 A	1.44 A	2.54 A	2.54 A	1.30 A	1.31 A	1.31 A	1.31 A
	6	2.30 T	1.57 A	1.89 A	1.90 A	3.12 A	3.12 A	1.79 A	1.80 A	1.81 A	1.81 A
	8	3.02 T	2.04 A	2.35 A	2.36 A	3.70 A	3.70 A	2.29 A	2.29 A	2.31 A	2.30 A
1000	4	1.83 T	1.83 T	1.83 T	1.83 T	2.80 A	2.80 A	1.60 A	1.57 A	1.60 A	1.60 A
	6	2.68 T	2.67 T	2.68 T	2.67 T	3.51 A	3.51 A	2.22 A	2.19 A	2.23 A	2.22 A
	8	3.52 T	3.51 T	3.52 T	3.51 T	4.21 A	4.21 A	2.84 A	2.81 A	2.86 A	2.85 A
1500	4					3.50 T	3.50 T	2.31 T	1.85 A	1.90 A	1.88 A
	6					4.55 T	4.55 T	3.27 T	2.58 A	2.65 A	2.63 A
	8					5.61 T	5.61 T	4.24 T	3.31 A	3.40 A	3.38 A
2000	4					3.72 T	3.72 T	2.52 T	2.53 T	2.55 T	2.55 T
	6					4.88 T	4.88 T	3.60 T	3.60 T	3.64 T	3.64 T
	8					6.04 T	6.04 T	4.67 T	4.67 T	4.73 T	4.72 T

[a]Direct insertion by the launch vehicle; no upper stage is used.
[b]T = Titan, A = Atlas, D = Delta, L = LLV3

Technology too heavy for Titan IV

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